Inactive JAN 30 1941 UNCLASSIFIED

J.O. RFC 222
R.A. 1422
RM No. L7A31

~~Confidential~~

ADVANCE COPY

NACA

RESEARCH MEMORANDUM

for the

Civil Aeronautics Administration, Department of Commerce

ESTIMATED FLYING QUALITIES OF THE MARTIN MODEL 202 AIRPLANE

By

Joseph Weil and Margaret Spear

Langley Memorial Aeronautical Laboratory
Langley Field, Va.

CLASSIFICATION CANCELLED

Authority NACA RF 28 61 Date 1/11/53

By DAFA 1/24/58 See

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON

JAN 24 1947

UNCLASSIFIED

NASA Technical Library
3 1176 01437 5001

UNCLASSIFIED

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

for the

Civil Aeronautics Administration, Department of Commerce

ESTIMATED FLYING QUALITIES OF THE MARTIN MODEL 202 AIRPLANE

By Joseph Weil and Margaret Spear

ADDENDUM

A conference on the Martin model 202 flying qualities estimations was held on January 20, 1947 with representatives of the Glenn L. Martin Company and the Civil Aeronautics Administration. At this time, it was learned that the positive range of the adjustable stabilizer on the prototype airplane had been changed from 4.4° to 2.5°. In addition, the maximum up and down elevator angles were increased by 5°. These changes should substantially improve the marginal elevator control problem in the landing condition at the foremost center-of-gravity position.

It was also learned that the Martin Company estimated the wing dihedral to be increased by about $1\frac{1}{4}^{\circ}$ in level flight as a result of structural deformation. No account was taken of the deformation in the analysis, therefore, it would be expected that the adverse dihedral effect cited will be somewhat alleviated.

UNCLASSIFIED

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

for the

Civil Aeronautics Administration, Department of Commerce

ESTIMATED FLYING QUALITIES OF THE MARTIN MODEL 202 AIRPLANE

By Joseph Weil and Margaret Spear

SUMMARY

The flying qualities of the Martin model 202 airplane have been estimated chiefly from the results of tests of an 0.0875-scale complete model with power made in the Wright Brothers tunnel at the Massachusetts Institute of Technology and from partial span wing and isolated vertical tail tests made in the Georgia Tech Nine-Foot Tunnel. These estimated handling qualities have been compared with existing Army-Navy and CAA requirements for stability and control.

The results of the analysis indicate that the Martin model 202 airplane will possess satisfactory handling qualities in all respects except possibly in the following:

The amount of elevator control available for landing or maneuvering in the landing condition is either marginal or insufficient when using the adjustable stabilizer linked to the flaps. Moreover, indications are that the longitudinal trim changes will be neither large nor appreciably worse with a fixed stabilizer than with the contemplated arrangement utilizing the adjustable stabilizer in an attempt to reduce the magnitude of the trim changes caused by flap deflection.

The available rudder control will probably enable landings to be made in cross winds at 90° to the path of only 11 percent of the stalling velocity for some conditions. This condition could probably be improved considerably; chiefly by using somewhat less than full flap deflection.

Considerable negative dihedral effect is probable in the landing and approach conditions which could make the airplane difficult if not dangerous to fly.

The aileron forces in abrupt rolls at cruising speeds are somewhat higher than the desired limits. Moreover, at the lower speeds the aileron forces are undesirably low or overbalanced. No change in the linkage arrangement of the linked-balancing tab would be likely to improve the control forces for one condition without having a detrimental effect on the other. However, it is shown that a spring-tab arrangement can be devised to provide reasonably satisfactory characteristics for all conditions.

INTRODUCTION

At the request of the Civil Aeronautics Administration, Department of Commerce, an estimate was made of the handling qualities of the Martin model 202 transport. This analysis was desired by the CAA as an advance indication of the flight characteristics to be anticipated for the prototype airplane. Availability of such knowledge was believed important from the standpoint of safety, facility in planning the flight-test program and subsequently in expediting the tests themselves.

It was originally planned to base the estimations on the results of complete model tests to be made in one of the Langley 7- by 10-foot tunnels. However, a rather extensive investigation by the Martin Company of a complete model of the Martin model 202 had already been made at the Wright Brothers tunnel at M.I.T. (See references 1 to 4.) In addition, detailed isolated vertical tail tests and tests of a partial span wing to obtain aileron characteristics had been made at the Georgia Tech nine-foot tunnel. (See references 5 and 6.) Therefore, although the investigations did not cover all of the points desired for a complete estimation of handling qualities, the time-saving element prompted the decision to use these data which were already available.

COEFFICIENTS AND SYMBOLS

The following coefficients and symbols appear in the text and figures:

C_h hinge-moment coefficient of a control surface $\left(H/qb\overline{c}^2\right)$

$C_{h_{\alpha_t}}$ rate of change of hinge-moment coefficient with tail angle of attack

C_{h_δ} rate of change of hinge-moment coefficient with control-surface deflection

$C_{h_{\delta_t}}$ rate of change of control-surface hinge-moment coefficient with tab deflection

C_L lift coefficient (Lift/qS)

T_c' effective thrust coefficient $\left(\frac{\text{effective thrust}}{qS}\right)$

δ control-surface deflection with respect to chord line, degrees

i_t stabilizer setting with respect to wing root chord line, degrees; positive when trailing edge is down

α angle of attack of wing root chord, degrees

α_t angle of attack of tail surface, degrees

β sideslip angle, degrees

g acceleration due to gravity (32.2 feet per second)

V_i indicated airspeed $\left(0.964\sqrt{\frac{W/S}{\rho_o C_L}}\right)$, miles per hour

n_p neutral-point location, percent mean aerodynamic chord

V_{S_L} stalling speed in the landing condition, power off, miles per hour

V_{S_G} stalling speed in the glide condition, power off, miles per hour

V_{S_p} stalling speed in the climb condition, 75 percent normal rated power, miles per hour

V_{S_A} stalling speed in the approach condition, 45 percent normal rated power, miles per hour

V_p design maneuvering speed (see reference 7)

V true airspeed, feet per second

$pb/2V$	wing-tip helix angle, radians
F_p	rudder pedal force, pounds
Fw	elevator or aileron wheel force, pounds
δ_w	wheel deflection, degrees
p	rolling velocity, radians per second
H	hinge moment of a control surface, pound-feet
q	dynamic pressure $\left(\rho V^2/2\right)$, pounds per square foot
b	wing span, feet
b'	(with subscripts) span of a control surface, feet
$\overline{c}$	root-mean-square chord of a control surface behind hinge line, feet
W	airplane gross weight, pounds
S	wing area, square feet
ρ	mass density of air, slugs per cubic foot
ρ_o	mass density of air at sea level (0.002378 slug per cubic foot)

Subscripts:

e	elevator
r	rudder
a	aileron
f	landing flap
e_{bt}	elevator linked balance tab
e_{tt}	elevator trim tab
r_{st}	rudder spring tab

r_{tt} rudder trim tab

a_{bt} aileron linked balance tab

a_{tt} aileron trim tab

Abbreviations:

Wm propellers windmilling

T.O.P. take-off power

N.R.P. normal-rated power

c.g. center of gravity

M.A.C. wing mean aerodynamic chord, feet

CAA Civil Aeronautics Administration

M.I.T. Massachusetts Institute of Technology

AIRPLANE AND MODELS

The Martin model 202 airplane is an all-metal, low-wing, twin-engine monoplane with full cantilever wing and tail surfaces. A three-view drawing of the airplane is presented in figure 1. Among the design features are a fully retractable tricycle type alighting gear with steerable nose wheel, double-slotted flaps interconnected by mechanical means to an adjustable stabilizer for the purpose of minimizing trim changes when the flaps are lowered or raised and a vane type (Van Zelm) aileron (see fig. 1) for which a smaller aileron can be used than is customary to obtain the same maximum rolling effectiveness, thus permitting the flap span to be increased.

The elevator and rudder can be aerodynamically balanced by an unsealed overhang and either a linked-balance tab or spring tab. The ailerons are aerodynamically balanced by a sealed overhang and a linked-balance tab.

A summary of the physical characteristics of the airplane furnished by the manufacturer is presented in tables I, II, and III.

The complete model tested at M.I.T. was a 0.0875-scale model with power. Details of the model are given in references 1 to 4.

A description of the 0.30-scale vertical tail model and the 0.25-scale partial span wing panel model tested at Georgia Tech can be found in references 5 and 6, respectively.

TESTS AND ANALYSIS

Test conditions.- Most of the tests in the Wright Brothers tunnel at M.I.T. were conducted at a dynamic pressure of 16.37 pounds per square foot which corresponded to an airspeed of about 80 miles per hour. The test effective Reynolds number was about 666,000.

The tests in the Georgia Tech wind tunnel of the vertical tail model and of the partial span wing model were made at a dynamic pressure of 25.58 pounds per square foot which corresponded to an airspeed of about 100 miles per hour. The test effective Reynolds numbers were about 3,920,000 and 2,758,000, respectively.

Model configurations and power conditions.- The various airplane flight conditions were simulated in the complete model tests by a suitable variation of model configuration and power condition. The conditions referred to repeatedly in this paper are summarized in the following table:

Airplane flight condition	Model configuration		
	Landing flaps	Landing gear	Power
Gliding	Retracted	Up	Propellers off
Cruising	Retracted	Up	75-percent N.R.P.
Climbing	Retracted	Up	75-percent N.R.P.
Approach	35°	Down	45-percent N.R.P.
Landing	55°	Down	Propellers windmilling

Methods of Analysis

The Martin model 202 airplane is a commercial transport and hence is required to meet the stability and control requirements of the CAA (reference 7). However, the flying qualities have been analyzed using the latest Army-Navy specifications for stability and control as a guide (reference 8 or 9). This was done primarily

because of the more specific nature of these requirements for class II airplanes (transport category) as compared to the rather general coverage of the CAA requirements. Where important differences exist between the two sets of specifications which have a critical bearing on the estimations suitable, reference is made in the text. It is to be noted that often the conditions for which model data were obtained did not correspond exactly to those specified in references 8 or 9. Instances where these differences have a pertinent bearing will also be brought out in the discussion.

The normal operating weight of the Martin model 202 transport is around 36,000 pounds $(W/S = 41.9)$. However, a majority of the power-on complete model tests were run using a thrust coefficient variation based on a weight of 29,000 pounds $(W/S = 33.7)$ and therefore it was often necessary to base estimates on this latter weight. All estimates were made for the mean operational or design center-of-gravity location unless otherwise specified. A weight-balance summary is given in table IV.

The flying qualities of the airplane have been estimated from the data reported in references 1 to 6 using methods similar to those outlined in references 10 to 13. It was assumed that the control surfaces are mass-balanced, that there is no friction or stretch in the control system and that there is no fabric distortion. The power-off stalling velocities of the airplane were based on values of $C_{L_{max}}$ which were obtained by extrapolating the complete model tunnel data.

No hinge-moment data were available from which the control forces of the elevator could be estimated directly. Values of $C_{h_{\alpha_t}}(0)$ and $C_{h_\delta}\left(-0.0037 q_t/q\right)$ were therefore estimated utilizing the lifting surface theory of reference 14 and the results of hinge-moment correlations summarized in reference 15. The hinge moments were then estimated using the q_t/q calculated from complete model pitching-moment data. The tab effectiveness $\left(C_{h_{\delta_t}} = -0.0047 q_t/q\right)$ which was used to determine the effects of a linked balance tab and trim tab was obtained using the methods outlined in reference 16. The linkage factor which was used to convert the surface hinge moment to wheel force was supplied by the manufacturer and is presented in figure 2.

In instances where only propellers-off data were available, the effect of the windmilling propellers on stability was estimated using the methods of reference 17.

The rotary derivatives which were used in estimating the dynamic stability characteristics of the airplane were calculated using the methods of references 18 to 20.

The rudder-control characteristics were computed using both complete model data (transferred to the stability axis) and the results of the isolated vertical tail tests. The angles of attack and tail loads required for trim were obtained from complete model tests. The rudder deflection required for trim and the rudder hinge moments were obtained from the isolated tail tests. It was possible to estimate the approximate effective aspect ratio of the vertical tail using the limited rudder data available from complete model tests. The isolated tail data were then corrected to this aspect ratio. The yawing moment due to the aileron deflection required for steady sideslips was accounted for. Calculations were made for only one of the several possible rudder-balancing arrangements, namely, an unpreloaded spring tab with a spring constant of 12 pounds per degree tab deflection. This spring strength was chosen primarily to provide acceptable forces in the critical asymmetric-power condition. A curve of rudder linkage factor against rudder deflection is presented in figure 3.

The aileron-control estimates were based on the results of the partial span wing tests made at Georgia Tech. The tunnel corrections were computed from reference 21 and it was found that the rolling-moment coefficients presented in reference 6 were about 8 percent too high. This was taken into consideration in the computations. Curves showing the nonlinear linked-balance tab deflections and linkage factor against aileron deflection are presented in figure 4.

ESTIMATED FLYING QUALITIES

The Army-Navy requirements are divided into four main sections in references 8 or 9, namely:

D - Longitudinal Stability and Control
E - Directional Stability and Control
F - Lateral Stability and Control
G - Stalling Characteristics

The items in the present paper are numbered to correspond with the requirements of references 8 or 9. Whenever a particular requirement was of such nature that an analysis was not deemed feasible using the available wind-tunnel data, it has been omitted.

The Martin model 202 transport falls into the category of class II aircraft in references 8 or 9.

Section D - Longitudinal Stability and Control

D-2 Static longitudinal stability.- The airplane will possess positive static longitudinal stability, elevator fixed, throughout the center-of-gravity range for all required flight conditions. (See fig. 5.) It should be noted that the neutral point curve for 75 percent normal-rated power (W/S = 33.7 pounds per square foot) actually is equivalent to approximately normal-rated power for the much more frequently encountered condition of W/S = 41.9 pounds per square foot. Inasmuch as the estimated value of $C_{h_{\alpha_t}} = 0$, the elevator-free neutral points should be identical with the elevator-fixed neutral points. It should also be noted, however, that in order to obtain satisfactory maneuvering gradients for an airplane of this size and speed, the elevator must be fairly closely balanced (obtained through the use of a linked balancing tab in this instance) and small changes in the hinge-moment parameters (from either nonlinearity, manufacturing dissimilarities or slight errors in the estimations) might easily cause the elevator-free neutral point to shift by 5 percent mean aerodynamic chord or more. This would most likely cause difficulties in the climbing condition (fig. 5) for in this condition the elevator-fixed static margin is a minimum while the tail contribution is considerable. The use of a spring tab would largely reduce the effects of any small changes in the control hinge moments on the stability because of the automatic compensation to these changes (repeatability) which are an inherent characteristic of spring-tab systems.

Application to CAA requirements.- The CAA requires elevator-free stability specifically in the cruising, climbing, approach and landing conditions. However, the requirements either are the same or somewhat less severe than those of D-2 so that the preceeding discussion is applicable. The CAA also concerns itself with the stability becoming so great that excessive control forces will be encountered in steady flight. However, generally if the elevator balance is designed to give satisfactory control-force gradients in turning flight and control forces in landing, then satisfactory characteristics will also be obtained in steady flight conditions.

D-3 Elevator control power.- (1) It will be possible to obtain steady flight over the entire speed range for most of the required conditions. (See fig. 6.) The elevator control for stalling the airplane in the landing condition (at the foremost center of gravity),

however, becomes marginal with the contemplated flap full down stabilizer setting of 4.4°. If the cruising stabilizer setting (-1.6°) were used, ample elevator would be available. (See fig. 6(c).)

(2) It should be no problem to obtain the positive limit load factor or maximum lift coefficient in a turn in cruising flight conditions with the present elevator control (fig. 7). For turns in the landing condition $\left(i_t = 4.4^\circ\right)$, however, the elevator available would probably be insufficient to perform the required maneuver at the foremost center of gravity through most of the speed range in view of the aforementioned marginal control for stalling in steady flight in this same condition.

(3) The amount of elevator estimated (from tests in the presence of a groundboard) to be required to hold the airplane off the ground at a speed of $1.05V_{S_L}$ (88 miles per hour, W/S = 41.9) for various arrangements is shown in table V.

It may be seen that the elevator control appears marginal for landing at the foremost center of gravity for the contemplated prototype stabilizer setting of 4.4°. (Maximum available up elevator is 30°.) With the stabilizer set at -1.6° there is unquestionably sufficient elevator for landing throughout the center-of-gravity range.

(5) Often a critical requirement for the adequacy of elevator control is the ability of the control surface to raise the nose wheel during a take off at a speed of 80 percent of V_{S_L} with the center of gravity most forward. No data were available, however, with which to estimate the take-off characteristics.

D-4 Elevator control forces.- (1) and (2) Estimated characteristics in steady turning flight were computed for the most forward center-of-gravity location at sea level (a measure of the maximum gradient) and with the center of gravity at the most rearward location for an altitude of 10,000 feet (a measure of the minimum gradient). For each of these conditions, computations were made for the balance tab locked and a linked-balance-tab ratio of 0.4 lag (a linkage variation from 0.3 lead to 0.8 lag is provided for on the prototype airplane).

For all conditions investigated changes in normal acceleration will be approximately proportional to the change in pilot applied control force. (See fig. 7.)

The gradients of elevator control force in steady turns at 270 miles per hour, as estimated from data obtained with the model

in the cruising configuration, are much too high in the tab locked condition. Satisfactory gradients (between 20 and 55 pounds per g, references 8 or 9) are indicated, however, with a linked balance tab ratio of 0.4 lag. Again, it must be stressed that minor changes in $C_{h_{\alpha_t}}$ and C_{h_δ} could considerably change the estimated characteristics. To illustrate this point computations were made to determine the effect of having slightly different hinge-moment parameters than were estimated. It was assumed that $C_{h_{\alpha_t}}$ and $C_{h_{\delta_t}}$ were changed by 0.0005 and C_{h_δ} was changed by -0.0005. Then, in a turn at 270 miles per hour at sea level at the foremost center of gravity the gradient is increased by about 27 pounds per g for the linked-balance-tab ratio of 0.4 lag. On the other hand, if a spring tab system were resorted to, an increase of only 4 pounds per g would be attained using a spring constant of 8 pounds per degree tab deflection. Thus if a linked-balance-tab arrangement can not be found which will be satisfactory for all conditions, a suitable spring tab or geared spring-tab system would probably furnish a satisfactory solution.

(6) The control forces required to hold the airplane off the ground at $1.05V_{S_L}$ for two possible stabilizer settings and balance arrangements with the airplane initially trimmed at $1.4V_{S_L}$ away from the ground are illustrated in table V. It is seen that with a 0.4-lag balance tab the use of either the fixed or adjustable stabilizer would result in a wheel force less than the limiting 50 pounds.

D-6 Longitudinal trim changes.- As previously mentioned the Martin model 202 has an adjustable stabilizer linked to the flaps which moves when the flap is changed in a manner which was hoped to minimize the trim changes caused by flap deflection. The following table summarizes the estimated trim changes with a linked adjustable stabilizer as well as for a fixed stabilizer (set for trim in cruising flight). It should be noted that a flap setting of 35° (approach) has been used in estimations requiring a landing setting (55°) because of the insufficiency of the test data at the latter setting. The trim changes on the airplane caused by full flap deflection will therefore probably be somewhat different than the estimations indicate.

Trim condition						Items changed	Change in control force (lb)		Change in δ_e for trim (deg)
No.	Speed (mph)	δ_f (deg)	Gear	i_t (deg)	Power		Tab ratio zero	0.4 lag tab	
1(a)	131	0	Up	-1.6	50 percent N.R.P.	Flaps 35°, gear down, variable stabilizer (i_t 2.2°)	8.7 push	4.3 push	1.7 down
1(b)	131	0	Up	-1.6	50 percent N.R.P.	Flaps 35°, gear down, fixed stabilizer	46.0 push	22.1 push	9.3 down
2	105	35	Down	2.2	50 percent N.R.P.	Power off	1.7 pull	0.9 pull	0.5 up
3	273	0	Up	-1.6	N.R.P.	Power off	3.7 push	1.5 push	0.2 down
4(a)	105	35	Down	2.2	Take-off	Flaps 0°, gear up, variable stabilizer (i_t -1.6°)	14.7 push	8.7 push	3.8 down
4(b)	105	35	Down	-1.6	Take-off	Flaps 0°, gear up, fixed stabilizer	13.6 pull	6.6 pull	3.6 up
5	105	35	Down	2.2	Power-off	Take-off power	5.2 push	2.5 push	1.5 down

Inspection of the table shows that the use of the 0.4 lag balance tab gives trim changes well under the specified 50-pound limit regardless of whether a linked adjustable or fixed stabilizer is incorporated. It is apparent that the adjustable stabilizer does

not always insure less trim change than a fixed stabilizer (compare conditions 1 and 4). In fact, under certain conditions the changes may be even greater with the adjustable stabilizer. This consideration, the lack of information as to the magnitude of trim changes with the flaps full down and, in addition, the aforementioned marginal elevator control associated with the adjustable stabilizer probably would make it of prime interest to check the relative merits of a fixed and adjustable stabilizer on the first flying article.

D-7 Longitudinal trimming device.- (2) The longitudinal trimming device is powerful enough to secure zero elevator control forces over the center-of-gravity range for the specified conditions. (See fig. 8.)

Section E - Directional Stability and Control

E-1 Dynamic stability.- The rudder fixed dynamic stability of the airplane was investigated for a cruising and a gliding flight condition. For the cruising condition (V_i = 240 miles per hour) the airplane will be spirally stable. The oscillatory stability in this condition is such that the period of the oscillation will be about 3 seconds and will damp to 1/2 amplitude in somewhat less than 1 cycle. In a gliding condition (V_i = 141 miles per hour), however, the airplane will be spirally unstable. But the spiral mode is such that the divergence will double in about 45 seconds so that the pilot should have no difficulty in controlling it. The period of the oscillation in the gliding condition is about 5 seconds and is also rather heavily damped, the time to damp to half amplitude again being appreciably less than 1 cycle.

E-2 Static directional stability.- (1) The airplane exhibits rudder fixed static directional stability for all flight conditions investigated. (See fig. 9.) It should be noted that the stability is high in the approach condition and particularly high in the landing condition. The angle of sideslip is roughly proportional to the rudder deflection from trim for all conditions.

(2) The critical condition for sideslip caused by rolling is encountered in gliding flight on the Martin Model 202 airplane. When gliding at V_i = 141 miles per hour in a 45° banked turn, the rudder-fixed static directional stability is such that the angle of sideslip caused by full aileron deflection is only 12.8° which is considerably less than the permissible 20°. (See fig. 10.)

(3) The rudder-force characteristics are such that with the rudder free the airplane will always tend to return to the trim condition with the wings level. (See fig. 9.) Although there are no actual force reversals shown, there are tendencies toward overbalance at the larger angles of sideslip. Moreover, overbalance may well occur on the airplane for some of these conditions if the basic limitations of the model hinge-moment data are considered. The data were insufficient, however, to establish whether or not rudder lock would occur at sideslip angles greater than 15° to 20°. If rudder lock is encountered on the airplane at the higher angles of sideslip, it may be desirable to utilize a larger dorsal fin in order to ameliorate the deficiency. As will be pointed out later on in the discussion calculations were made for only a spring tab system. With a linked-balance-tab arrangement or with a combination of the two arrangements even greater tendencies toward overbalancing would be encountered.

(4) The single-engine condition investigated and referred to in this and subsequent sections pertaining to asymmetric power conditions is for a climbing condition at a speed of about $1.2V_{S_G}$. The thrust simulated in the right engine (left propeller windmilling) corresponded to about rated power at W/S of 41.9. Although this condition is sometimes more stringent and at other times less stringent than those specified in references 7 to 9, similarity was generally close enough that the one condition investigated could be used throughout. The data were insufficient to establish whether or not the airplane could be balanced directionally in steady straight flight for the aforementioned condition with the rudder free and trim tab neutral. (See fig. 11.)

The amount of pitching moment resulting from sideslip although not mentioned in references 8 or 9 is generally considered to be of interest to airline pilots. This airplane will probably meet requirement II-H of reference 22, for it is estimated that, for all conditions shown on figure 9, less than 1° elevator movement is needed to maintain longitudinal trim when the rudder is moved 5° in either direction from its trim position at zero bank.

E-3 Rudder control power.- (1) From the conditions investigated and shown in figure 9, it appears safe to assume that the rudder will be sufficiently powerful to trim the airplane in all probable steady symmetric flight conditions with the wings level.

(2) The only complete model yaw data available for the landing condition was at a lift coefficient corresponding to a speed of about 100 miles per hour. Using these data, it is estimated that,

a landing at about 80 miles per hour in a cross-wind at 16 miles per hour (20 percent V_{S_L}) at 90° to the flight path would require trim to be attainable at $\beta = 12°$ sideslip. It can be seen, figure 9(e), that only about 6° left and 11° right sideslip can be held with the contemplated maximum rudder deflection (±25°). The trimmable sideslip range could probably be increased by using less than the full down flap setting. (See fig. 9(d).) A slight increase would also be obtainable by increasing the maximum rudder throw to ±30°, but the resulting increased severity of rudder lock tendencies might not warrant the change. No data were available with which to estimate the rudder control during take-offs.

(4) For the asymmetric power condition investigated, figure 11, it is seen that about 22° right rudder is needed to hold zero sideslip. Rudder may not be available to meet the actual requirement E-3-4, however, inasmuch as full take-off power was not simulated. Moreover, the directional stability would be expected to be somewhat greater with the flaps in the take-off setting (10° to 15°).

Application to CAA requirements.- Sufficient rudder control is probably available to execute 20° banked turns with or against the inoperative engine from a steady climb at a speed of $1.4V_{S_G}$ with maximum continuous power being applied to the operating engine. For although the thrust coefficient simulated in the model tests analyzed corresponded to only about rated power at $C_L = 1.0$ ($W/S = 41.9$) at the C_L for $1.4V_{S_G}$ (about 0.7) the power represented would be even greater than rated power for $W/S = 33.7$.

No data were available for single engine operation in the approach condition, but because of the large directional stability, it is extremely doubtful whether heading changes of 15° against the inoperative engine could be achieved from trim with the wings level.

(5) It is estimated that only about 4.5° of right rudder will be needed to overcome adverse aileron yaw during an abrupt full right aileron roll from a 45° banked turn in the gliding condition at $1.4V_{S_G}$.

E-4 Rudder pedal forces.- As has been previously mentioned, no particular difficulty is expected to be encountered in obtaining satisfactory control forces because of the wide variety of linkage arrangements and springs of different strengths with which the

prototype airplane can be equipped. As an example calculations using one logical arrangement were made. This arrangement made use of an unpreloaded spring tab with a spring constant of 12 pounds per degree tab deflection.

(1) If the means suggested are used to increase the trimmable sideslip range, it is shown in figures 9(d) and 9(e) that the pedal forces will be considerably less than 180 pounds for the required cross-wind landing. Only about 20 pounds of rudder pedal force will be required to counteract the adverse aileron yaw in condition E-3-5.

(2) The pedal force required to hold zero sideslip in the asymmetric power condition investigated was roughly 120 pounds with the trim tab set for zero pedal force in the symmetric climb condition. (See figs. 9(c) and 11.) The actual pedal force for the required condition would probably be somewhat larger as has been previously pointed out.

E-6 Directional trimming device.- (2) If a spring tab system is used on the rudder, some thought should be given to the use of a separate tab for trimming and balancing purposes. This is advisable because of the possibility of reduced tab effectiveness for balancing when most of the unstalled lift range of the tab is used up for trimming. Also in spring tab systems the tab hinge moments cause the tab to blow back against the spring requiring extra effort on the part of the pilot in the form of repeated trim jack adjustments. Because this airplane only attains moderately high speeds and has not been designed for violent maneuvering, a combination spring and trim tab might be acceptable.

(3a) The directional trimming device is easily capable of reducing the rudder pedal force to zero in the gliding and climbing flight conditions with the wings level. (See figs. 9(b) and (c).)

(b) About 16.6° left tab (maximum deflection 20°) is necessary to trim in the asymmetric power condition investigated. Inasmuch as in this instance the required condition is slightly less severe, sufficient trim tab should be available to meet it.

Section F - Lateral Stability and Control

F-2 Static lateral stability.- (1) The airplane will probably be laterally statically stable with both fixed or free ailerons in all flap-up conditions. (See figs. 9(a) to 9(c).) However, because of the effects of double-slotted flap deflection, considerable

negative effective dihedral is indicated in the computed approach and landing conditions (figs. 9(d), and 9(e)). The curves of rolling-moment coefficient versus yaw angle for these latter conditions with the rudder fixed at 0° showed either a small amount of stability or neutral stability, but because of the high directional stability present in these conditions and the large change in rolling moment with rudder deflection, the slope of the curve of total aileron deflection for trim against β indicates appreciable instability. It should be remembered that because of the limitations of the available data the flap-down estimates were made at speeds corresponding to angles of attack which would be considerably lower than would be normally used in the approach and landing conditions. At these low angles of attack the flaps were stalled on the model casting some doubt on the applicability of the data to the higher C_L range or even the low C_L range at the higher Reynolds numbers of the airplane. Nevertheless, unless the rolling- and yawing-moment characteristics are much changed at the larger angles of attack, even the rudder-fixed dihedral effect will be negative because of the greater adverse effect of power in the approach condition and because of the reduced tail contribution to positive dihedral effect at the larger angles of attack in both the approach and landing conditions. If the dihedral effect indicated in figures 9(d) and 9(e) persist on the airplane at the higher lift coefficients, it is believed trouble may be encountered on several scores. In making instrument approaches at fairly low altitudes the pilot brackets a slender range leg, generally using the rudder alone to accomplish this. The rate of spiral divergence would be high and would be continually initiated and aggravated by use of the rudder in obtaining and maintaining headings. In a final landing approach it would be possible to have the rudder fully deflected when attempting to maintain a ground track while trying to raise a wing dropped by a gust. The $pb/2V$ available for leveling the wings in this condition would give rolling velocities far below those now desired by airline pilots.

(2) The small effective dihedral coupled with a rather large directional stability insures that the rolling moment caused by sideslip in a rudder-fixed aileron roll will never be large enough to cause a reversal of rolling velocity because of aileron yaw.

(3) The variation of side force with angle of sideslip will be such that right bank accompanies right steady sideslip and vice versa for all conditions investigated. (See fig. 9.)

F-3 Aileron control power.- (1) There are no differences between the Van Zelm ailerons used on the model 202 and conventional ailerons which would cause the airplane to roll in the wrong direction immediately after an abrupt aileron deflection.

(3) For all conditions investigated, figure 12, the rolling velocity will vary smoothly with aileron deflection and be approximately proportional to the amount of the deflection.

(4) The helix angle obtained with maximum aileron deflection will be approximately equal to or greater than the required $pb/2V = 0.070$ for all conditions except in the gliding flight condition at a speed of 128 miles per hour (fig. 12) where a maximum value of $pb/2V = 0.064$ is obtainable. The values of $pb/2V$, however, were obtained from the rolling-moment data of reference 6 and contain an arbitrary correction factor of 20 percent to account for the effects of adverse yaw and wing twist. This correction factor is believed to be conservative inasmuch as the wing twist will probably not be large at the speeds reached and the discussions of E-2-2 and F-2-2 indicate the adverse yaw effects will be rather small on this airplane.

(5) The value of pb will probably be considerably greater than 10 feet per second at $1.1V_{S_L}$ when maximum aileron deflection is used.

(6) Sufficient aileron control is available to secure lateral trim in the asymmetric power flight conditions investigated, about two-thirds of the available aileron being required. (See fig. 11.) Inasmuch as the ailerons remain effective up to full throw, it is probable that requirement F-3-6 could be met for any probable asymmetric power condition likely to be encountered flaps up.

(7) The ailerons are effective enough to obtain a $pb/2V$ of 0.05 per 100° of wheel throw up to at least a speed of 240 miles per hour (approximately $0.8V_{max}$). (See fig. 12.)

F-4 Aileron forces.- (1) The aileron control-force characteristics in rolling maneuvers and steady sideslips are not always of sufficient gradient to return the control to trim position when cogniscence is taken of the allowable frictional limit of 6 pounds. (See figs. 9 and 12.) In fact, there is actual overbalancing indicated for an abrupt full roll in the gliding flight condition at the lower speed investigated (fig. 12).

(2) The estimated value of $0.8V_{max}$ in level flight is approximately 240 miles per hour. At this speed it would require a wheel force of about 150 pounds, (fig. 12) to attain the required $pb/2V = 0.07$. The force is considerably greater than the allowable 80 pounds. Moreover, because of the overbalancing tendencies at the lower speeds, it does not appear feasible to reduce the high-speed force by a change in the linked balance tab deflection rate.

Application to CAA requirements.- Information relative to CAA rolling requirements can be gleaned from the section on strength requirements in reference 7. There it is stated that full aileron deflection is required only up to the design maneuvering speed (V_p = approximately 170 miles per hour for this airplane) and that when the design cruising speed is in excess of V_p the rate of roll required at the design cruising speed be not less than that obtained using full aileron deflection at V_p. Assuming a maximum $pb/2V$ of about 0.07 at V_p, it is apparent that only a $\frac{pb}{2V}=\frac{170}{240}\times 0.07 = 0.0495$ will be required at 240 miles per hour. The control force required to obtain this $pb/2V$ is about 95 pounds, (fig. 12) still over the 80-pound limit, but possibly tolerable. Nevertheless, overbalancing is indicated at lower speeds and thus it was decided to investigate the characteristics with a spring-tab system.

Control forces and $pb/2V$ were estimated for a spring-tab system with individual unpreloaded spring units with a spring strength of 2 pounds per degree tab deflection at zero aileron deflection. Calculations were made for a spring tab the same size as the present linked tab and for a spring tab of 50 percent increased effectiveness. Both tabs were assumed aerodynamically balanced with a deflection range of ±15°. The curve of δ_a versus δ_w shown in figure 4 was modified so that for the same maximum wheel deflection (120°) no reduction in maximum aileron deflection will be obtained when the spring tab is fully deflected. It can be seen, figure 13, that in the gliding condition at V_i = 240 miles per hour the wheel force for a $pb/2V$ of 0.0495 is reduced to about 60 pounds with the tab of 50 percent increased effectiveness. Moreover, although in the gliding condition at V_i = 127 miles per hour there still exists a slight reduction in control force for increased aileron deflection in the range of total deflections of between 22° and 35°, the objectionable overbalance has been eliminated.

(3) The aileron control force for trim at zero sideslip in the asymmetric power flight condition investigated will be of the order of 20 pounds. This is far less than the limiting 80 pounds. (See fig. 11.)

F-6 Lateral trimming devices.- (3a) The lateral trimming device will be powerful enough to reduce the aileron force to zero in the gliding and climbing conditions at all required speeds. (See figs. 9(b) and (c).)

(3b) Approximately maximum available trim tab (27.5°) will be needed to reduce the aileron control force to zero at zero sideslip for the asymmetric power condition investigated. However, it should also be noted that the specific condition outlined for F-6-3b should also be met inasmuch as it is of less severity than that shown on figure 11.

Section G - Stalling Characteristics

G-2 Stall warning.- Because of differences in scale which will undoubtedly cause the stall characteristics to differ on the airplane and model, the discussion will be of a brief qualitative nature based chiefly on the tuft studies and discussion of reference 3.

Good stall warning will be realized in the landing condition $(\delta_f = 55°)$ inasmuch as the root section unmistakably stalls first. In the approach condition the stall also begins over the inboard portion of the wing but spreads outboard over the ailerons more than in the landing condition. For the gliding condition $(\delta_f = 0°)$ the stall started at the inboard trailing edge and gradually spread forward and outward. The addition of power $(\delta_f = 0°)$ delayed the stall in the nacelle region. For both flap-up conditions, a good portion of the aileron region was stalled before $C_{L_{max}}$ was reached.

There generally will be a fairly marked increase in the rearward travel of the control column as stall approaches. (See fig. 6.) This is especially true in the landing condition.

G-3 Prevention of the complete stall.- The tuft sketches showed that a good portion of the ailerons were stalled before $C_{L_{max}}$ was reached in the flap-up flight conditions. The data of reference 6, however, indicate that considerable aileron effectiveness exists up to the stall. In the landing condition, the ailerons remained unstalled above the angle of attack for $C_{L_{max}}$. As has been shown previously, the available elevator to stall with the stabilizer set in the flap-full-down position was marginal. However, for less positive stabilizer settings in the landing condition and in other conditions it should be possible to prevent or recover from the complete stall by the normal use of the controls when corrective action is taken immediately after the stall warning occurs.

G-4 Differential stalling of the wings.- Any differential stalling on the airplane will depend critically on the amount of

asymmetry in the actual airplane. Model data showed that in flap-down conditions the right wing panel stalled considerably earlier than the left panel. It was also shown, however, that the stall occurred on the inboard portion of the wing so that the rolling or yawing moments incurred would probably be controllable.

CONCLUDING REMARKS

The results of the analysis based on the available wind-tunnel data indicate that the Martin Model 202 airplane will probably possess satisfactory handling qualities in all respects except possibly in the following.

1. The amount of elevator control available for landing or maneuvering in the landing condition is either marginal or insufficient when using the adjustable stabilizer linked to the flaps. Moreover, indications are that the longitudinal trim changes are neither large nor appreciably worse with a fixed stabilizer than with the contemplated arrangement utilizing the adjustable stabilizer in an attempt to reduce the magnitude of the trim changes caused by flap deflection.

2. Indications are that the available rudder control will enable landings to be made in cross winds at 90° to the path of only 11 percent of the stalling velocity for some conditions. This condition could be improved; chiefly by using somewhat less than full flap deflection.

3. Considerable negative effective dihedral is probable in the landing and approach conditions which could make the airplane difficult if not dangerous to fly.

4. The aileron forces in abrupt rolls at cruising speeds are somewhat higher than the desired limits. Moreover, at the lower speeds the aileron forces are undesirably low or overbalanced. No change in the linkage arrangement of the linked balancing tab would be likely to improve the control forces for one condition without having a detrimental effect on the other. However, a spring-tab

arrangement can be designed that will provide reasonably satisfactory characteristics for all conditions.

Langley Memorial Aeronautical Laboratory
National Advisory Committee for Aeronautics
Langley Field, Va.

Joseph Weil
Aeronautical Engineer

Margaret F. Spear
Engineering Aide

Approved: Hartley A. Soulé
for Hartley A. Soulé
Chief of Stability Research Division

CMH

REFERENCES

1. Hess, Weiner F., and Coughlin, K. J.: Wind Tunnel Results of Model 202 - First Series, 8.75% Scale Model. Aero. Sec. Rep. No. 217, The Glenn L. Martin Co., March 1946.

2. Taylor, Robert D.: Wind Tunnel Tests on Glenn L. Martin Company Model M-202, Second Series. M.I.T. Wind Tunnel Rep. No. 714, Feb. 1946.

3. Cooper, L. G.: Wind Tunnel Results of Model 202 Third Series Tests - 8.75% Model. Effect of Running Propellers. Aero. Sec. Rep. No. 225, The Glenn L. Martin Co.; March 1946.

4. Carell, Marguerite E.: Wind Tunnel Tests on Glenn L. Martin Company Model M-202, Fourth Series. M.I.T. Wind Tunnel Rep. No. 723, Feb. 1946.

5. Pope, A. Y.: Wind Tunnel Tests of the Model 202 Rudder. Georgia Tech. Rep. No. E-69, Daniel Guggenheim School of Aeronautics, Feb. 1946.

6. Ducoffe, C. L., Moseley, W. C., Jr., and Whitaker, Charles L., Jr.: Wind Tunnel Tests of the Model 202 Aileron. Georgia Tech Rep. No. E-66, Daniel Guggenheim School of Aeronautics, Dec. 1945.

7. Anon.: Airplane Airworthiness - Transport Categories. Pt. 04 of Civil Air Regulations, Civil Aero. Board, U. S. Dept. Commerce, Nov. 9, 1945.

8. Anon.: Specification for Stability and Control Characteristics of Airplanes. SR-119A, Bur. Aero., April 7, 1945.

9. Anon.: Stability and Control Characteristics of Airplanes. AAF Specification No. R-1815-A, April 7, 1945.

10. Kayten, Gerald G.: Analysis of Wind-Tunnel Stability and Control Tests in Terms of Flying Qualities of Full-Scale Airplanes. NACA ARR No. 3J22, 1943.

11. Zimmerman, Charles H.: An Analysis of Lateral Stability in Power-Off Flight with Charts for Use in Design. NACA Rep. No. 589, 1937.

12. Swanson, Robert S., and Toll, Thomas A.: Estimation of Stick Forces from Wind-Tunnel Aileron Data. NACA ARR No. 3J29, 1943.

13. Schuldenfrei, Marvin: Some Notes on the Determination of the Stick-Fixed Neutral Point from Wind-Tunnel Data. NACA RB No. 3120, 1943.

14. Swanson, Robert S., and Crandall, Stewart M.: Lifting-Surface-Theory Aspect-Ratio Corrections to the Lift and Hinge-Moment Parameters for Full-Span Elevators on Horizontal Tail Surfaces. NACA TN No. 1175, 1947.

15. Langley Research Department (Compiled by Thomas A. Toll): Summary of Lateral-Control Research. NACA TN No. , 1947.

16. Crandall, Stewart M., and Murray, Harry E.: Analysis of Available Data on the Effects of Tabs on Control-Surface Hinge Moments. NACA TN No. 1049, 1946.

17. Ribner, Herbert S.: Notes on the Propeller and Slipstream in Relation to Stability. NACA ARR No. L4I12, 1944.

18. Pearson, Henry A., and Jones, Robert T.: Theoretical Stability and Control Characteristics of Wings with Various Amounts of Taper and Twist. NACA Rep. No. 635, 1938.

19. Bamber, Millard J.: Effect of Some Present-Day Airplane Design Trends on Requirements for Lateral Stability. NACA TN No. 814, 1941.

20. Swanson, Robert S., and Priddy, E. LaVerne: Lifting-Surface-Theory Values of the Damping in Roll and of the Parameter Used in Estimating Aileron Stick Forces. NACA ARR No L5F23, 1945.

21. Sivells, James C., and Deters, Owen J.: Jet-Boundary and Plan-Form Corrections for Partial-Span Models with Reflection Plane, End Plate, or No End Plate in a Closed Circular Wind Tunnel. NACA TN No. 1077, 1946.

22. Gilruth, R. R.: Requirements for Satisfactory Flying Qualities of Airplanes. NACA ACR, April 1941.

TABLE I

PHYSICAL CHARACTERISTICS OF THE MARTIN MODEL 202 AIRPLANE

Type	Commercial transport
Engines	
Manufacturer's designation	Pratt & Whitney R-2800-CA3
Ratings:	
Normal power	1700 bhp at 2600 rpm at sea level 1700 bhp at 2600 rpm at 7,000 ft 1450 bhp at 2600 rpm at 18,500 ft
Take-off power[1]	2100 bhp at 2800 rpm at sea level
Supercharger type	Single stage, two speed
Propeller gear ratio	0.450
Propeller:	
Type	Hamilton Standard reversible pitch
Diameter, ft	13.08
Blade design	2H17B3-48R
Number of blades	3
Activity factor (per blade)	168
Side-force factor	132
Landing gear:	
Tricycle (nose-wheel) type	

[1]Water injection rating of 2400 bhp simulated in all take-off power complete model testing.

NATIONAL ADVISORY
COMMITTEE FOR AERONAUTICS

TABLE II

AIRPLANE WING AND TAIL-SURFACE DATA

	Wing	Horizontal tail	Vertical tail
Area, sq ft	860	275.1	[1]118.0
Span, ft	92.75	36.47	14.297
Aspect ratio	10	4.84	1.732
Taper ratio	2.75	2.5	2.215
[2]Dihedral, deg	3	8	
Sweepback, quarter chord line, deg	0		
Root section	W-16 (19 percent) modified	65-013 modified	65-011 modified
Tip section	W-16 (15 percent) modified	65-010 modified	65-011 modified
Angle of incidence at root, deg	[3]3	[4]varies from -1.6° to 4.4°	0
Angle of incidence at break, deg	[3]3	↓	0
Angle of incidence at tip, deg	[3]3	↓	0
M.A.C., ft	10.02	8.069	8.767
Theoretical root chord, ft	13.67	10.833	11.490
Theoretical tip chord, ft	4.97	4.333	5.186

[1]Includes no dorsal fin area.
[2]Dihedral measured at quarter chord line.
[3]Angle of incidence measured with respect to fuselage base line.
[4]Angle of incidence measured with respect to wing chord line.

TABLE III

AIRPLANE CONTROL-SURFACE DATA

	Aileron	Elevators	Rudder	Wing[5] flaps	Fuselage flaps[6]
Percent span	[1]12.2	90	100	63.4	9.9
Area, behind hinge line, sq ft	16.6	24.5	39.67	150	30
Balance area, sq ft	4.32	7.63	15.12		
Percent chord behind hinge line	23.0	20.0	34.0	25.67	
Mean chord, behind hinge line, ft	1.437	1.50	2.796		
Distance to 1/4 tail M.A.C. from 1/4 wing M.A.C., ft		35.46	33.85		
Control deflection, deg	30 up 15 down	[3]26 up 16 down [4]30 up 15 down	25 right 25 left		
Trim-tab area, sq ft	1.33	3.43	3.93		
Span, ft	4	7.1	6.50		
Tab deflection, deg	12 up 16 down	±15	±20		
Balance tab area, sq ft	(2)	(2)	(2)		
Tab deflection, deg	9 up 12 down	.8 lag .3 lead	±10		

[1]One side
[2]Same tab used for trimming and balancing
[3]Flaps up
[4]Flaps down ($\delta_f = 55^\circ$)
[5]Double slotted flaps
[6]Split flaps

Flap deflections, deg (corresponding powers)

For landing . . . . . 45-55 (power off)
For take-off . . . . . 10-15 (2100-2400 hp)
Approach . . . . . 30-35 (765 hp)
All other conditions . . . . . Flaps retracted

NATIONAL ADVISORY
COMMITTEE FOR AERONAUTICS

TABLE IV

AIRPLANE WEIGHT-BALANCE SUMMARY

Condition	Gross weight (lb)	Center of gravity, wheels up			Center of gravity, wheels down		
		(in. behind fuselage station 0)[1]	(percent M.A.C.)	(in. above thrust line)	(in. behind fuselage station 0)[1]	(percent M.A.C.)	(in. above thrust line)
Design	36,000	448	27.4	12.4	450	29.2	
Foremost c.g.		434	14.4	10.5	436	16.0	6.9
Rearmost c.g.		452	31.0	14.3	452	31.0	10.7

[1]Fuselage station 0 is 100 inches forward of nose.

NATIONAL ADVISORY
COMMITTEE FOR AERONAUTICS

TABLE V

LANDING CHARACTERISTICS

$\left[V_i = 1.05V_{S_L} = 88 \text{ mph}, W/S = 41.9\right]$

Condition		16 percent c.g. (foremost)				31 percent c.g. (rearmost)			
i_t (deg)	Balance tab	$\delta_{e_{req}}$ (deg)	$\delta_{e_{tt}}$ (deg)	$\delta_{e_{bt}}$ (deg)	F_w (lb)	$\delta_{e_{req}}$ (deg)	$\delta_{e_{tt}}$ (deg)	$\delta_{e_{bt}}$ (deg)	F_w (lb)
4.4	Locked	29 up	6.4 down	0	57.0 pull	14.8 up	4.8 down	0	25.2 pull
↓	0.4 lag	29 up	3.1 down	11.6 down	28.0 pull	14.8 up	3.7 down	5.9 down	8.5 pull
-1.6	Locked	13.3 up	2.4 up	0	40.9 pull	3.0 up	6.7 up	0	27.3 pull
↓	0.4 lag	13.3 up	1.2 up	5.3 down	19.7 pull	3.0 up	3.3 up	1.2 down	13.3 pull

NATIONAL ADVISORY
COMMITTEE FOR AERONAUTICS

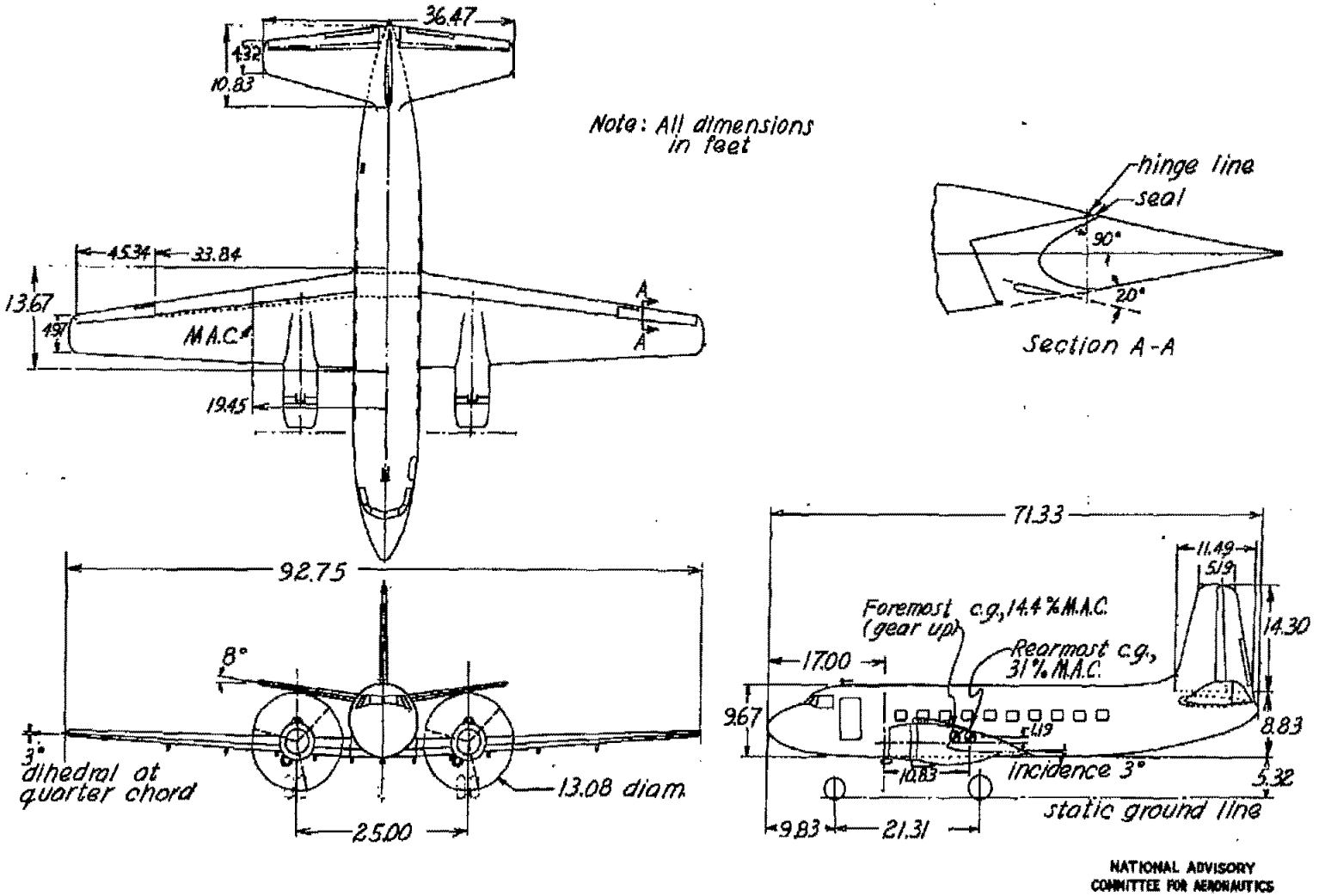


Figure 1.-Three-view drawing of the Martin Model 202 airplane.

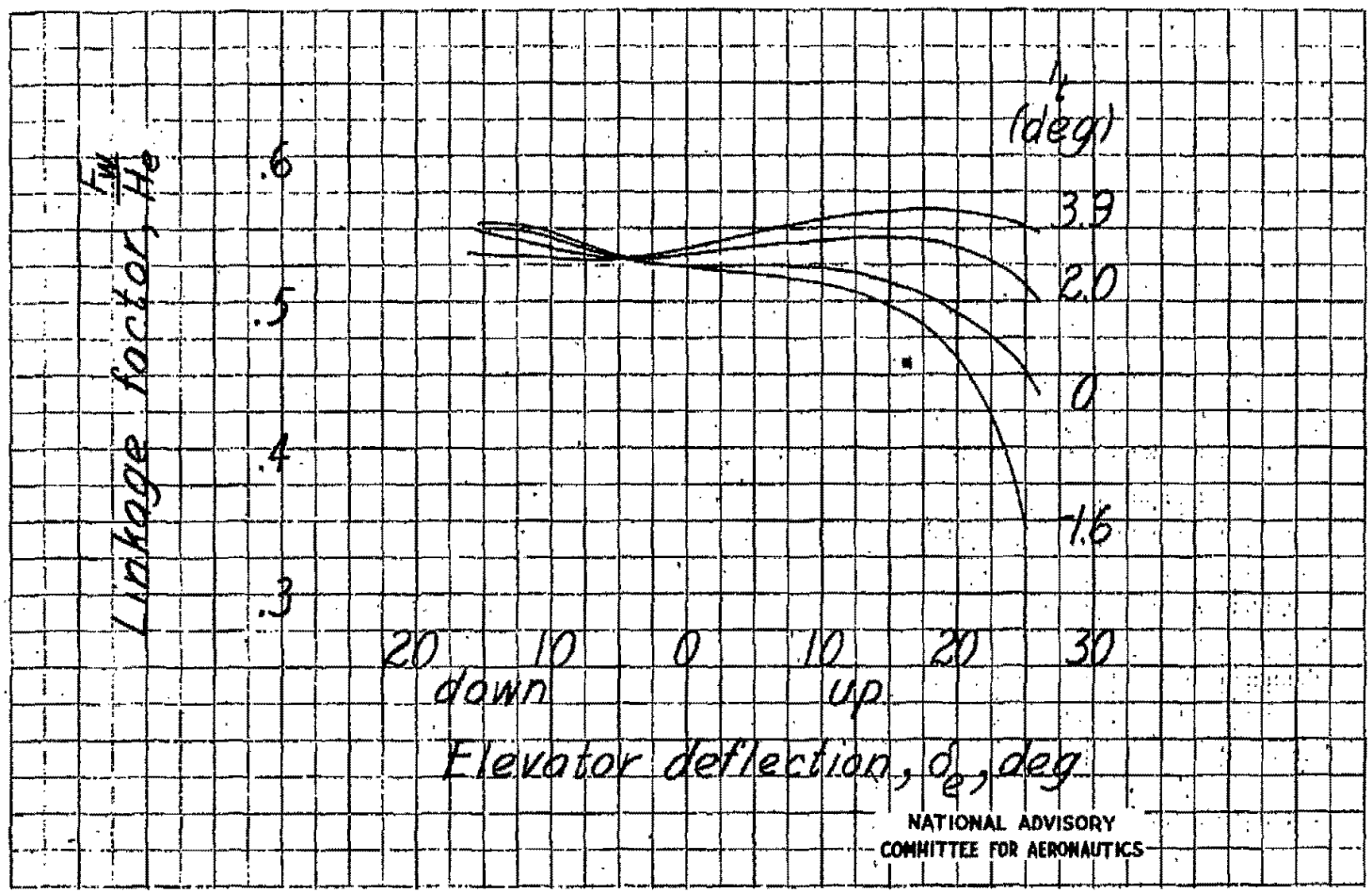


Figure 2.- Variation of elevator linkage factor with elevator deflection on the Martin Model 202 airplane.

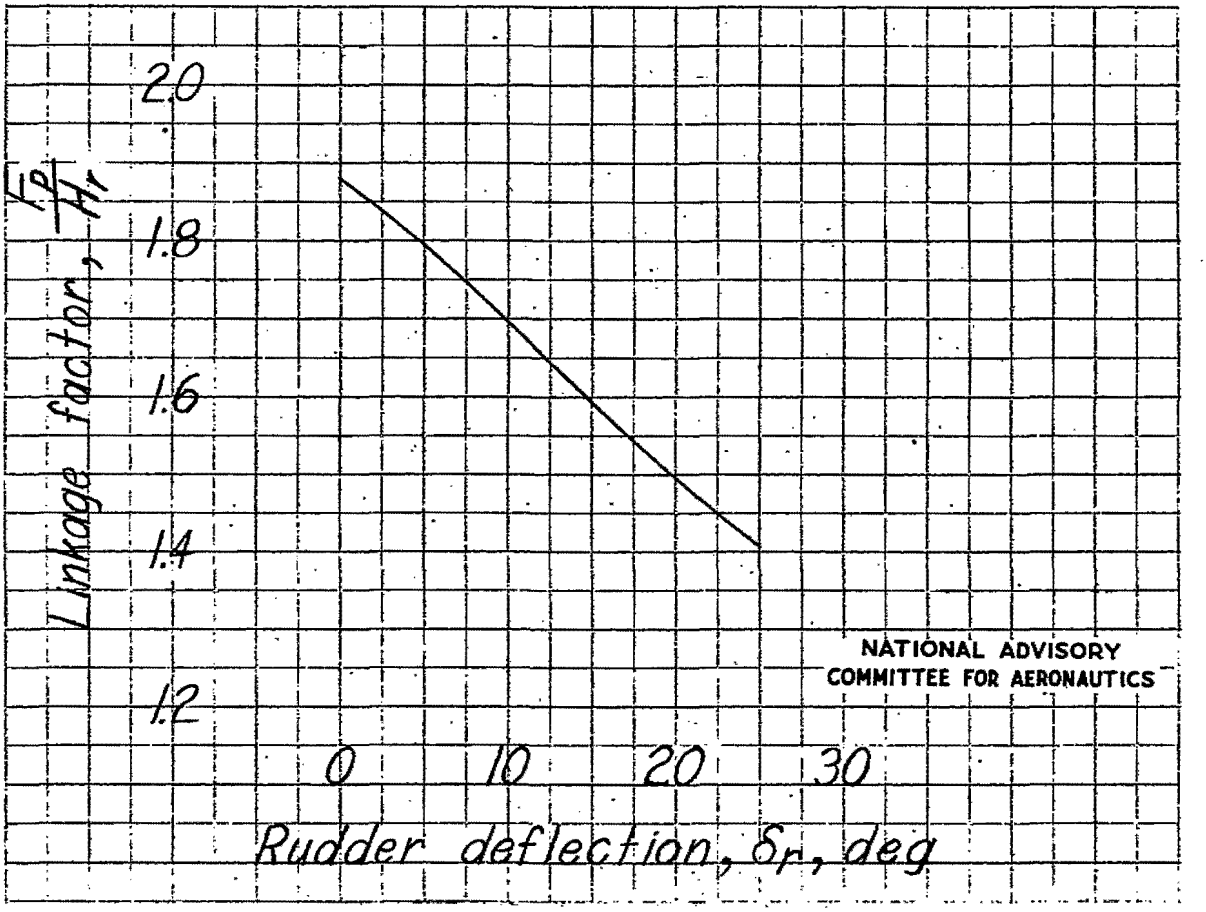


Figure 3.- Variation of rudder linkage factor with rudder deflection on the Martin Model 202 airplane. Spring tab locked, no load.

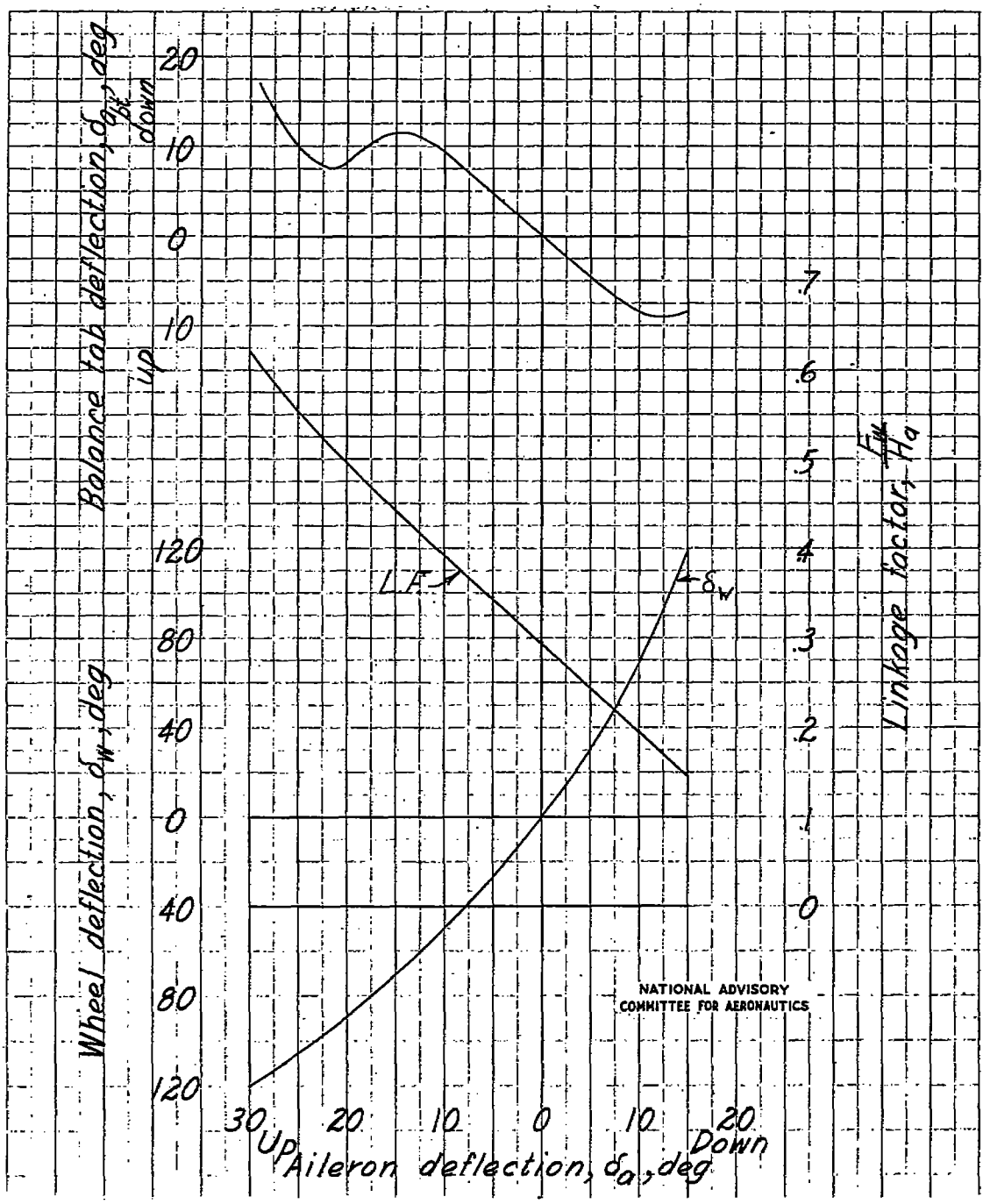


Figure 4.- Variation of aileron linkage factor, wheel deflection, and linked balance tab deflection with aileron deflection on the Martin Model 202 airplane.

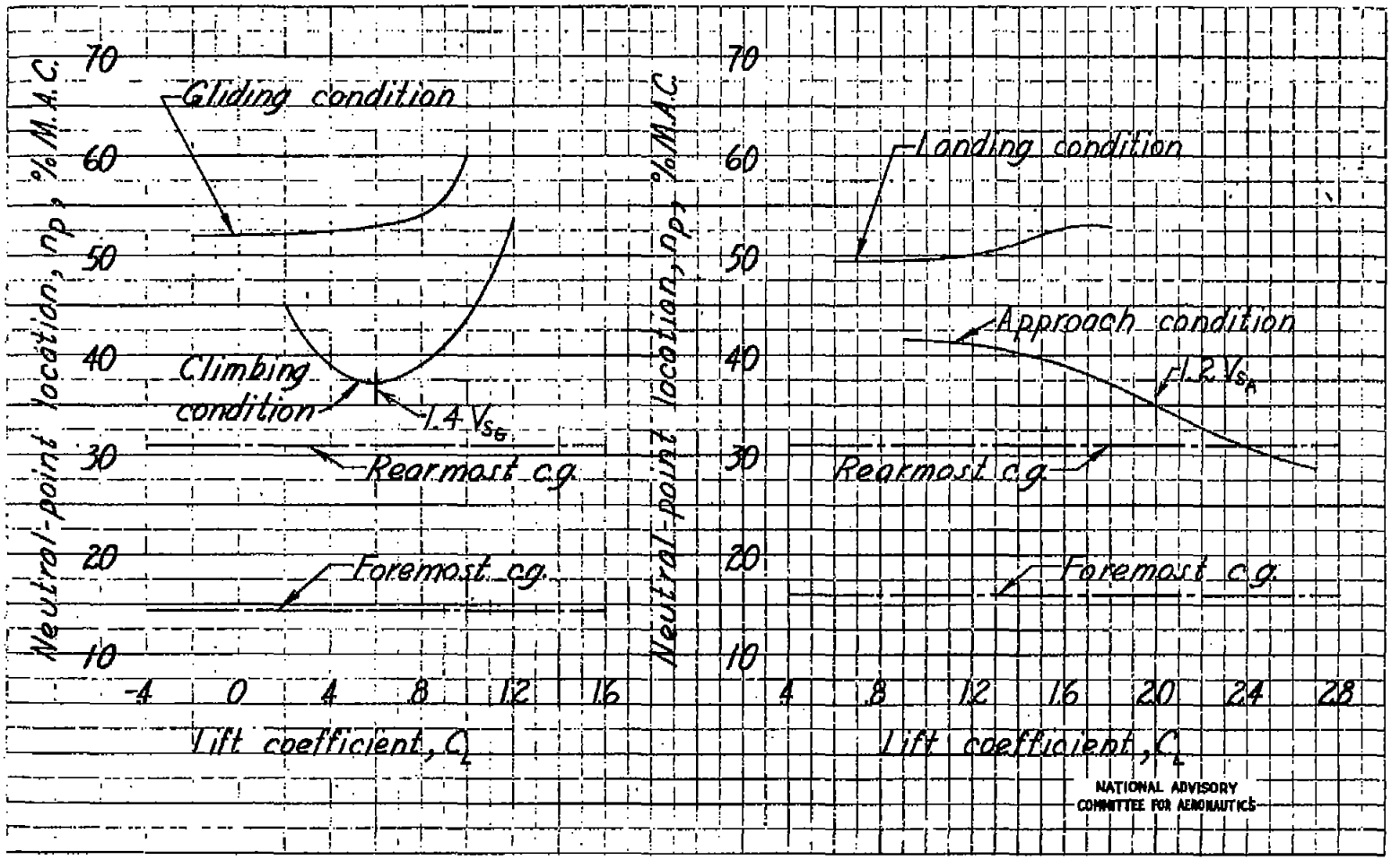


Figure 5.- Estimated elevator-fixed neutral points of the Martin Model 202 airplane. W/S = 33.7 lb/sq ft.

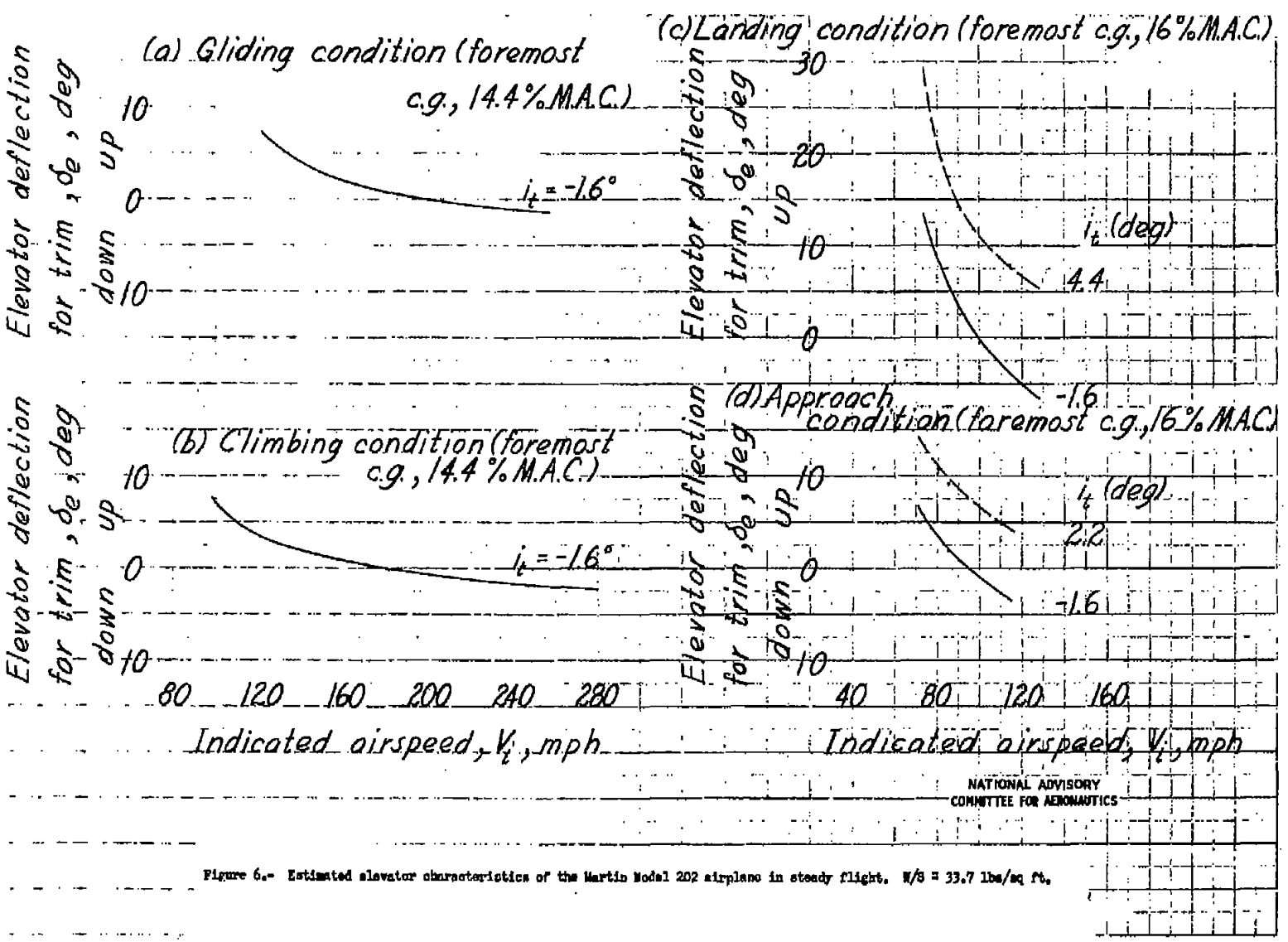


Figure 6.- Estimated elevator characteristics of the Martin Model 202 airplane in steady flight. W/S = 33.7 lbs/sq ft.

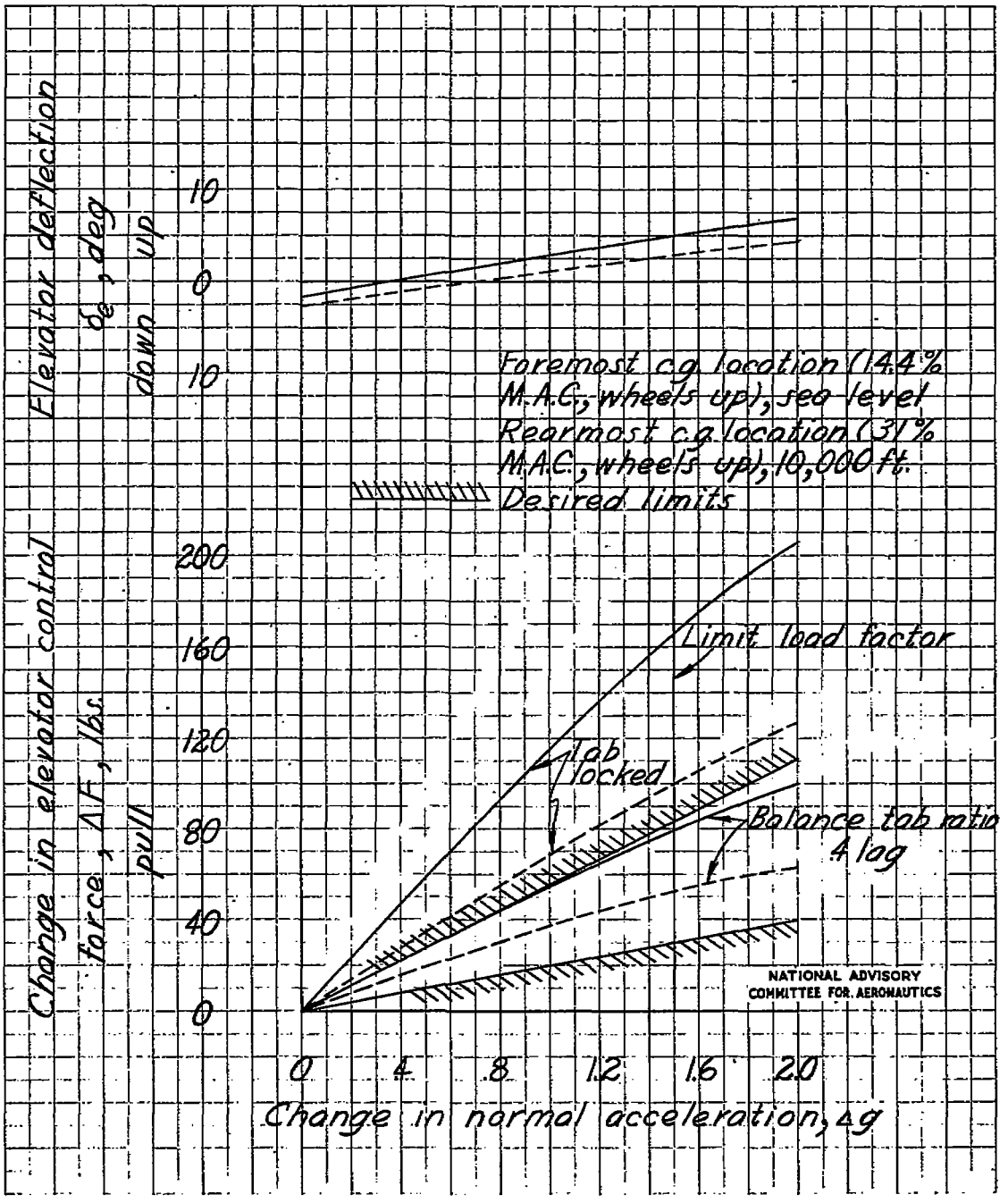


Figure 7.- Estimated elevator-control characteristics in accelerated flight for the Martin Model 202 airplane. W/S = 33.7 lb/sq ft; V = 270 mph; i_t = -1.6°.

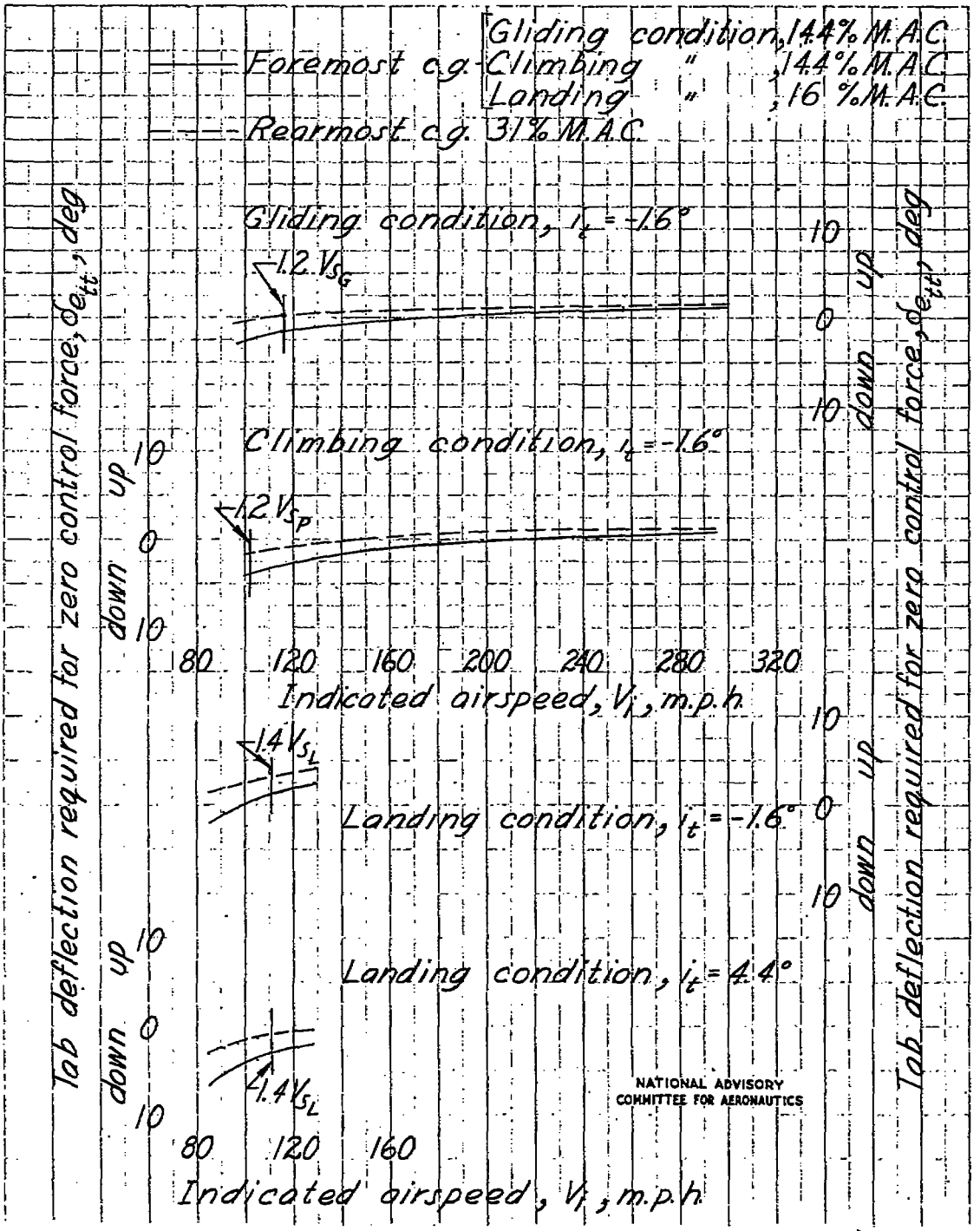
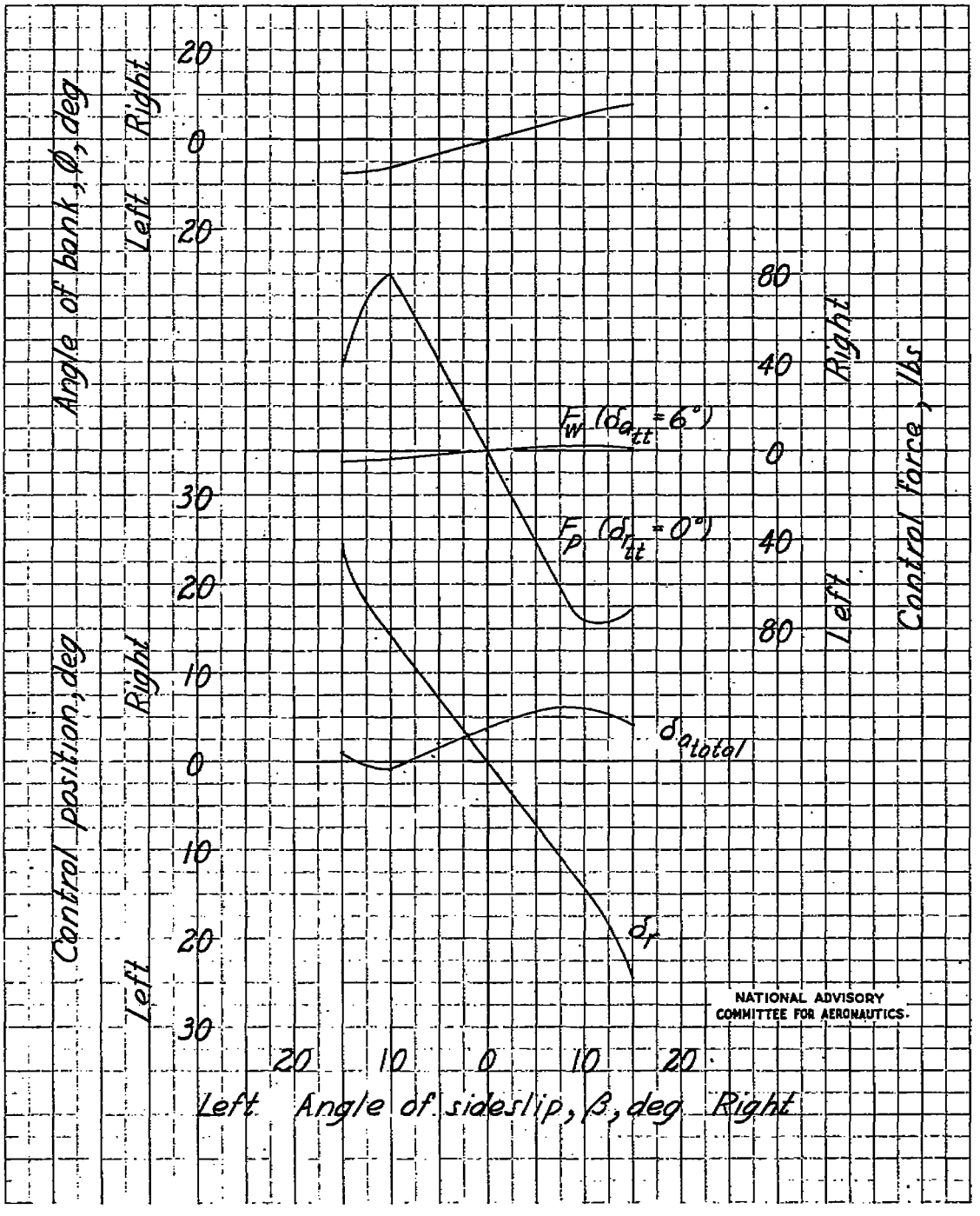
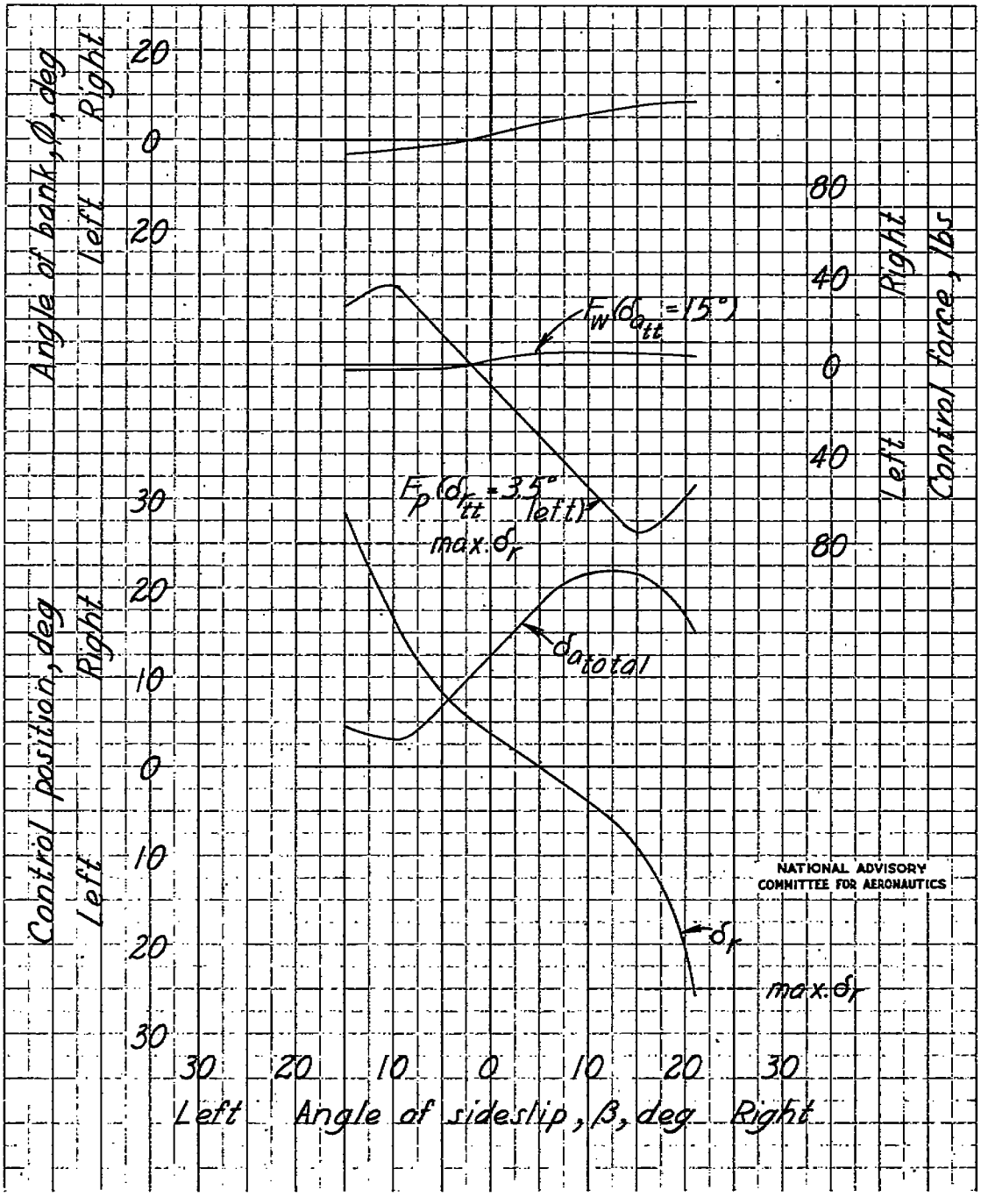


Figure 8.- Estimated trim tab deflection required for zero elevator control force on the Martin Model 202 airplane. Linked balance tab deflection 0.4 lag. W/S = 33.7 lb/sq ft.



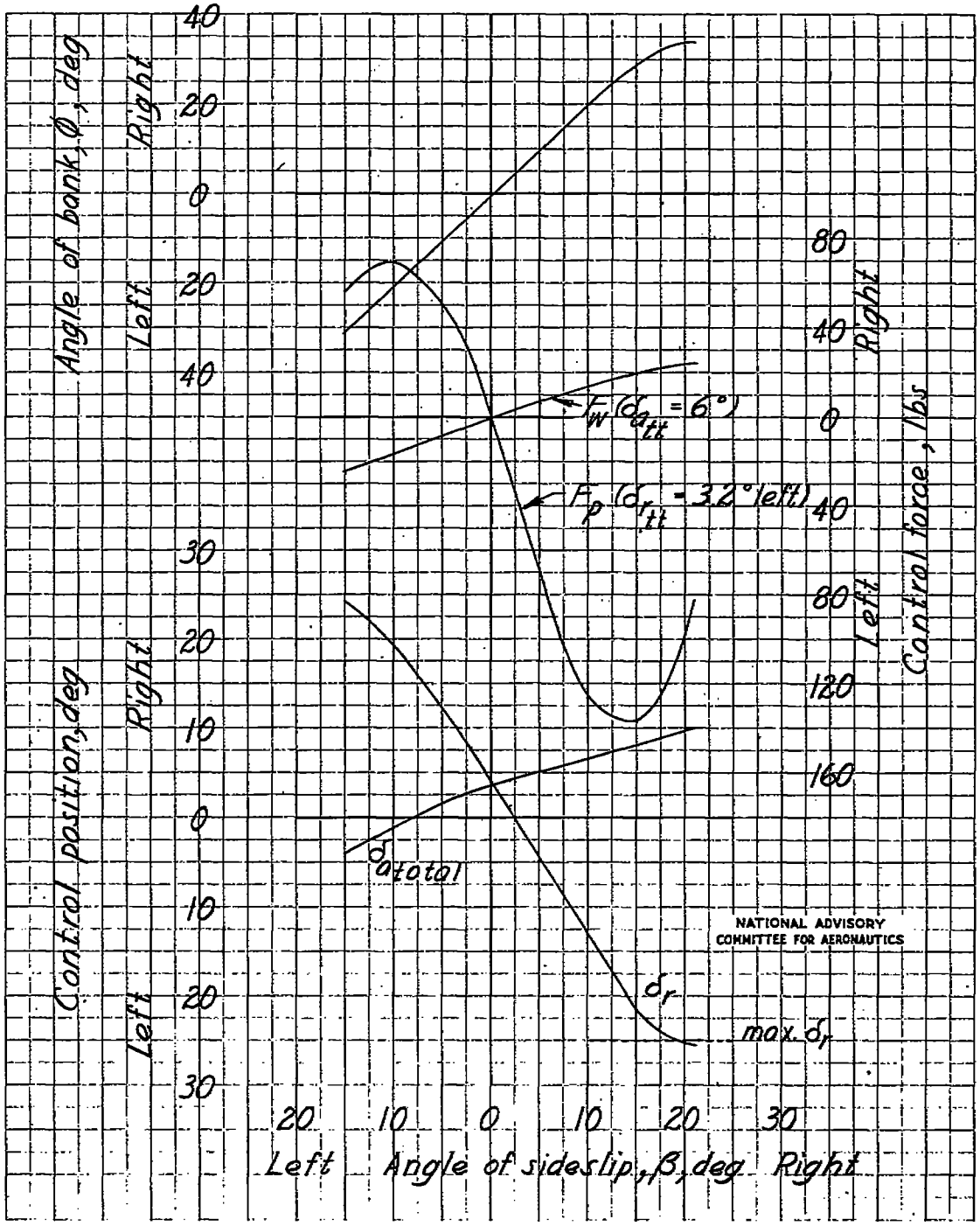
(a) Gliding condition, V_i = 127.2 mph

Figure 9.- Estimated steady sideslip characteristics of the Martin Model 202 airplane. W/S = 33.7 lbs/sq ft.



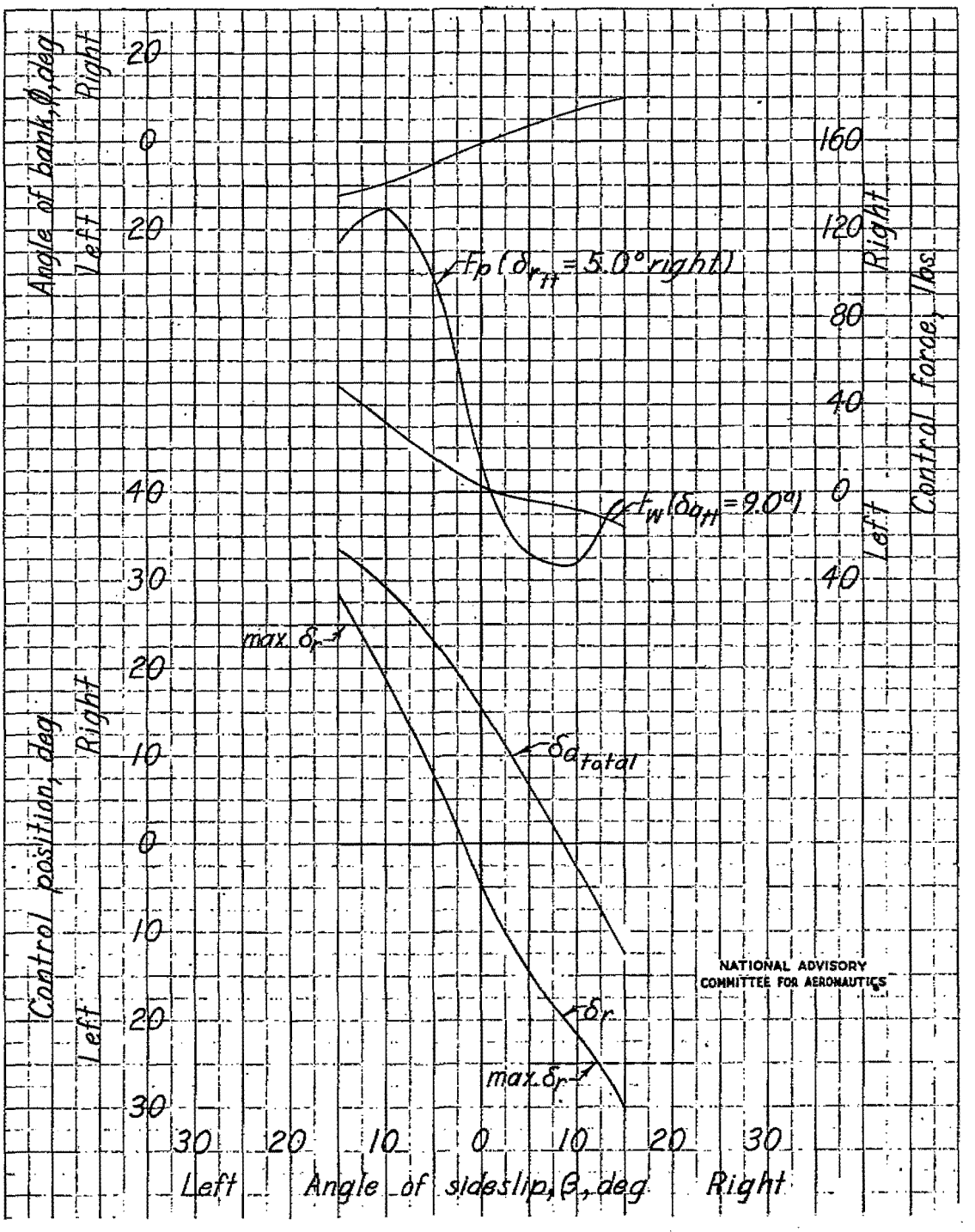
(b) Climbing condition, V_i = 101 mph

Figure 9.- Continued.



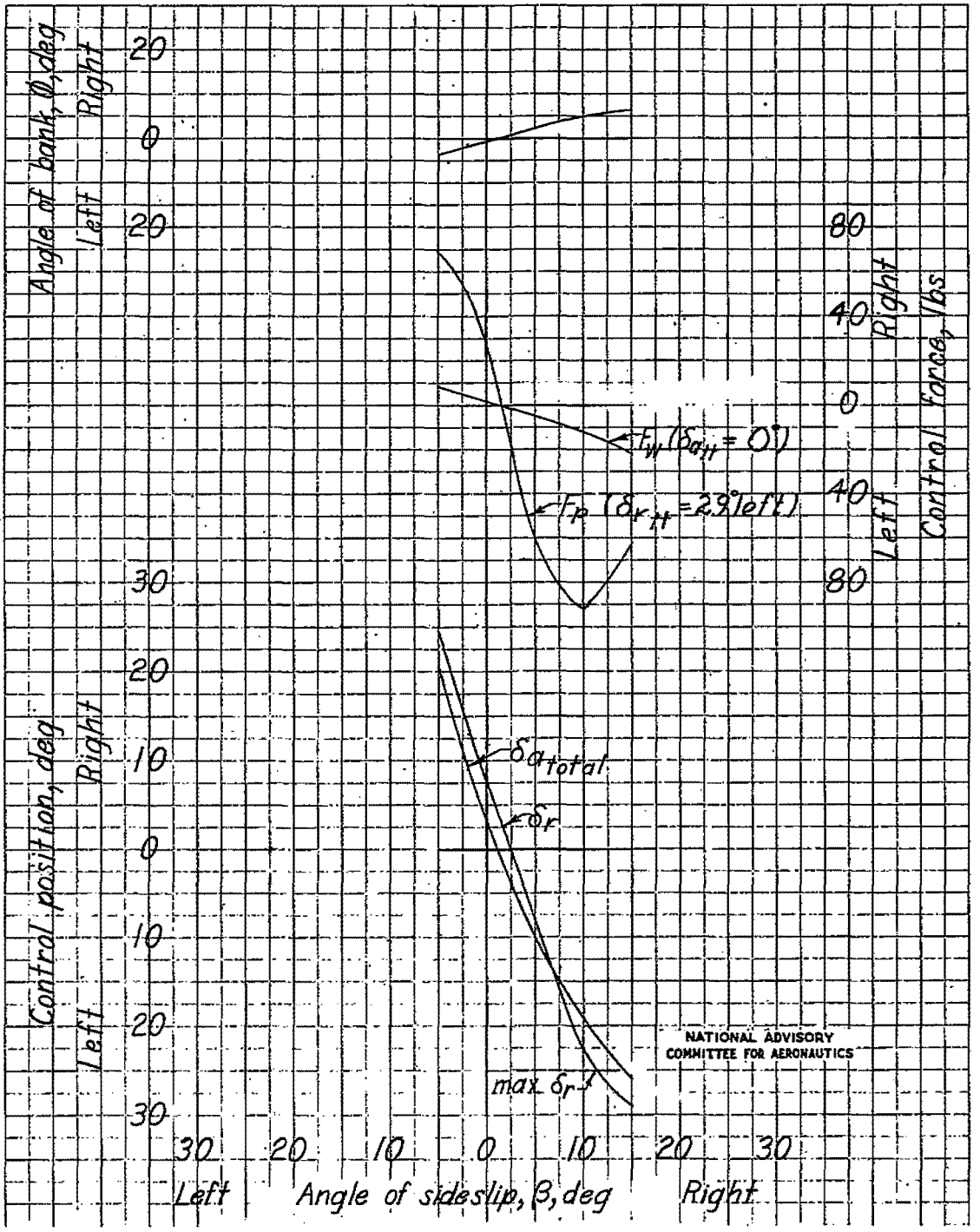
(c) Cruising condition, V_i = 239 mph

Figure 9.- Continued.



(d) Approach condition, V_i = 115 mph

Figure 9.- Continued.



(e) Landing condition, V_I = 100 mph

Figure 9.- Concluded.

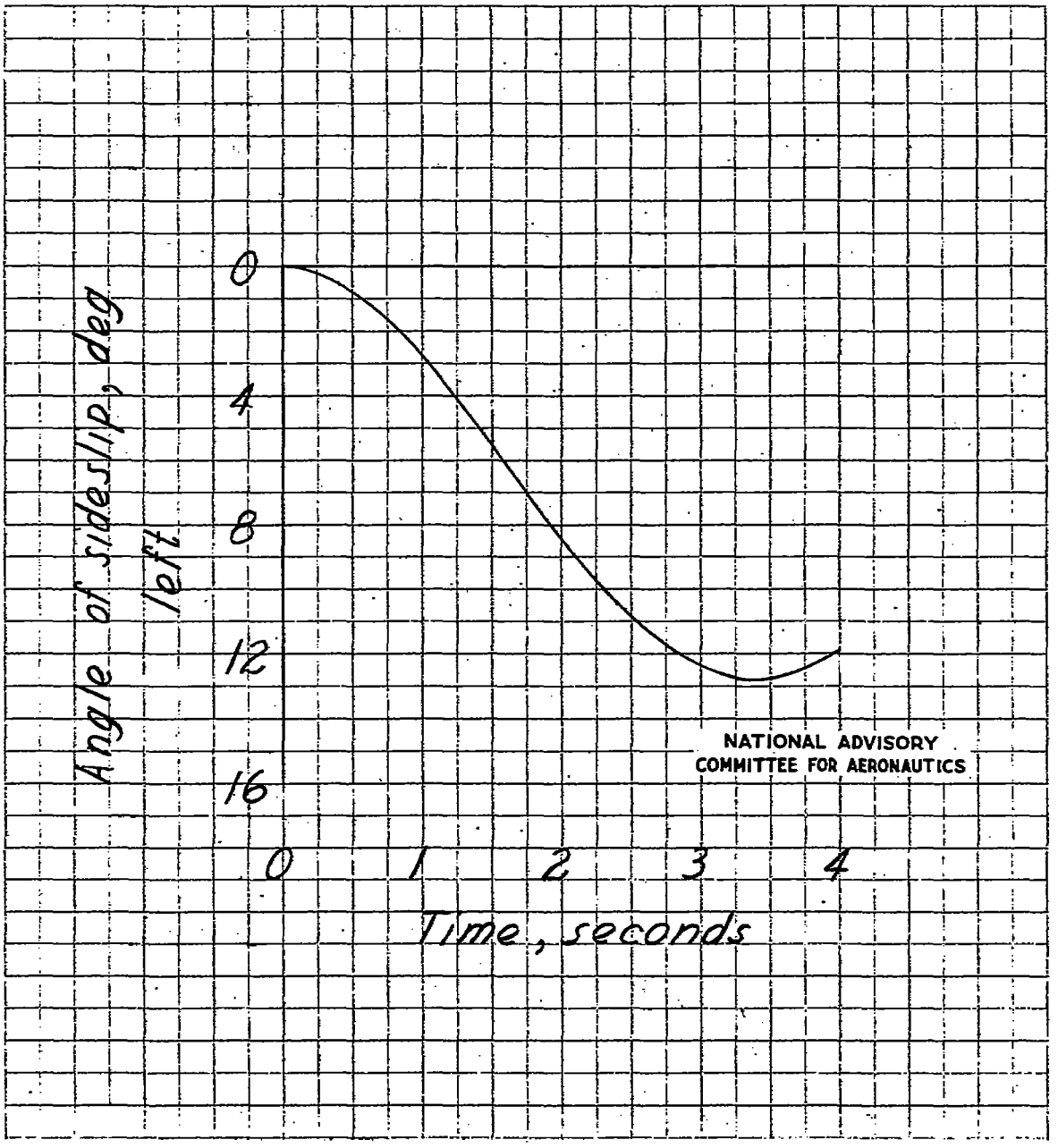


Figure 10.- Estimated variation of sideslip angle with time during an abrupt rudder-fixed aileron roll out of a steady 45° banked turn on the Martin Model 202 airplane. Gliding condition, V_i = 141 mph, full right aileron.

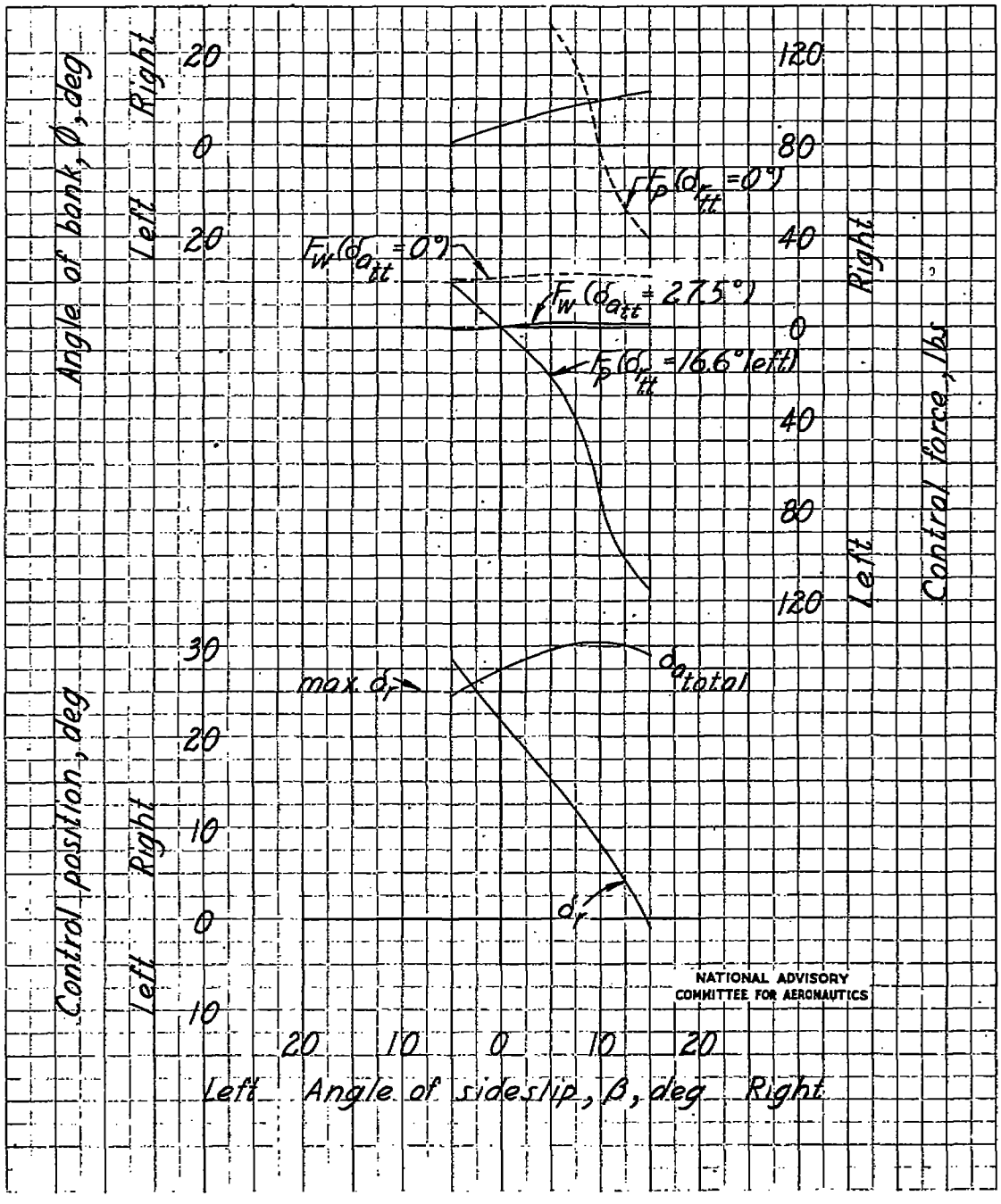


Figure 11.- Estimated single-engine characteristics in steady sideslips of the Martin Model 202 airplane. Left propeller windmilling; right engine $T_c' = 0.193$; $\delta_f = 0°$, $V_i = 115$ mph.

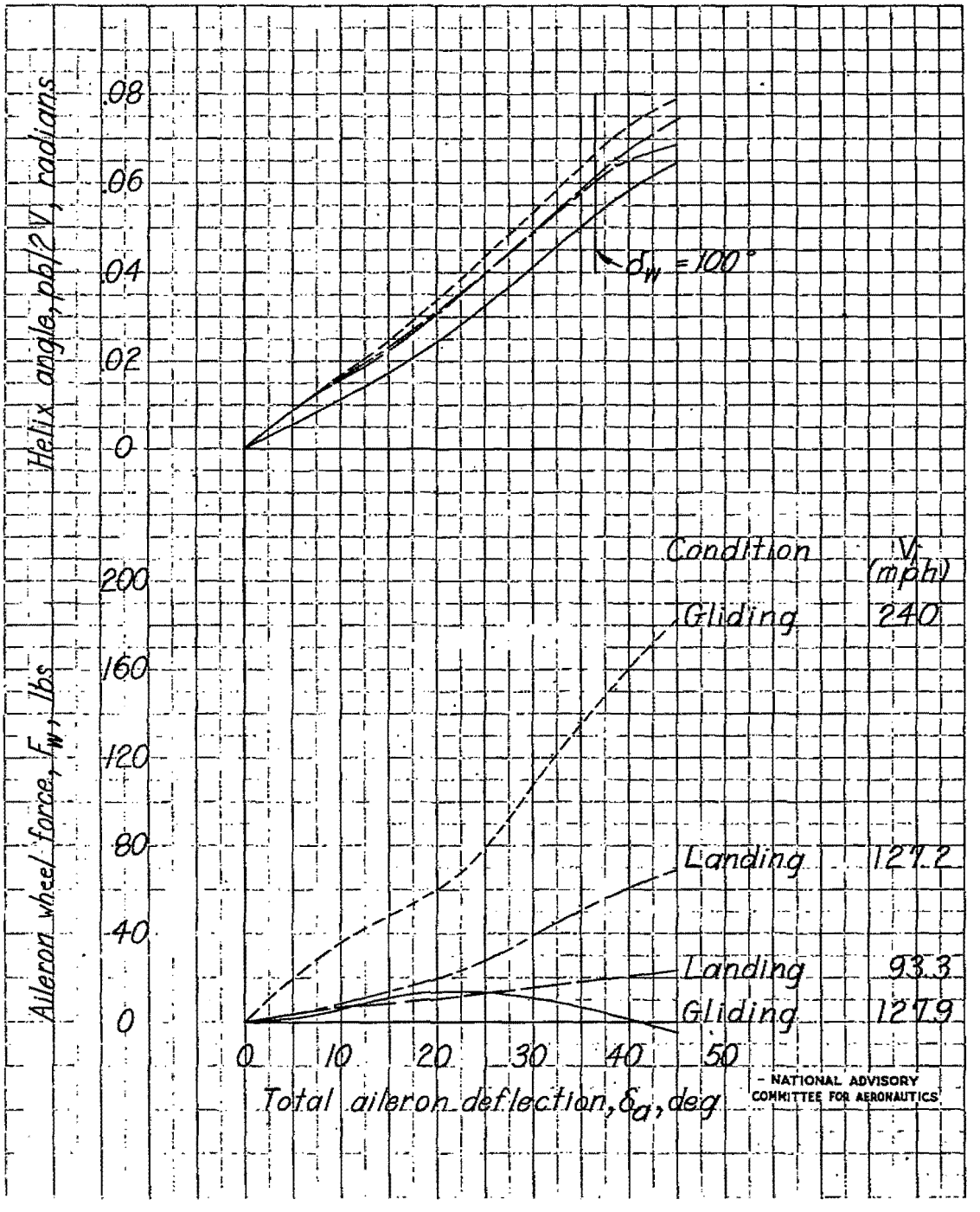


Figure 12.- Estimated variation of helix angle, and aileron wheel force with total aileron deflection for various conditions on the Martin Model 202 airplane. W/S = 41.86 lbs/sq ft.

NASA Technical Library
3 1176 01437 5001